FINAL REPORT

FOR

DESIGN AND FABRICATION

OF A

DEPLOYABLE LARGE AREA SOLAR CELL ARRAY

SUPPORTING STRUCTURE

RYAN

REPORT NO. 20869-3

28 APRIL 1967

Contract No. 951107

This work was performed for the Jet Propulsion Laboratory, California Institute of Technology, pursuant to a subcontract issued under Prime Contract NAS 7-100 between the California Institute of Technology and the United States of America represented by the National Aeronautics and Space Administration

RYAN AERONAUTICAL COMPANY • 8650 BALBOA AVENUE • SAN DIEGO, CALIFORNIA

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1.0 INTRODUCTION

This report summarizes the results of a three-phase study, design, and fabrication program conducted by Ryan Aeronautical Company for the Jet Propulsion Laboratory under JPL Contract 951107.

The purpose of the program has been to develop the concepts and technology required for designing and fabricating a deployable large area solar cell array structure using a minimum shroud volume.

During the program, a four-element 200 square foot rollup solar array structure was designed, and a 50 square foot prototype element was fabricated.

Phase I, Study, and Phase II, Design, have been previously reported in detail. This report summarizes the entire program and describes the fabrication of the prototype structure.

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2.0 SUMMARY

The development, design, and fabrication of the Deployable Large Area Solar Cell Structure have been accomplished in accordance with the requirements of JPL Contract 951107. The contract established hypothetical structural and environmental requirements similar to a 1969 Mars launch and encounter using chemical systems and an Atlas/Centaur launch vehicle.

The array concept is compatible to areas of from 150 to 400 square feet, with 200 square feet assumed as a target size for concept studies.

In Phase I a roller drum-extendable beam concept (Figure 1) was selected because of its inherent adaptability to growth requirements. A second advantage is the good self-damping qualities of the stowed substrate which minimize dynamic deflection problems.

In Phase II a four-element 200 square foot deployable array structure was designed to a total weight objective of 0.6 pound per square foot including 0.3 pound per square foot for the deployment system, structure and substrate and 0.3 pound per square foot for the cells, filters, modular wiring and secondary cabling.

In Phase III a 50 square foot prototype element has been fabricated to demonstrate feasibility of the design concept and deployment system. The structural materials and systems used are compatible with environmental and flight requirements including sterilization, launch, deployment, cruise, course correction maneuver, retraction, retromaneuver, and redeployment.

The prototype structure is suitable for functional and environmental testing in accordance with contract requirements.

The following is a summary of structural and thermal characteristics of the design as compared to contract requirements. The structure weight is greater than contract requirements, but reflects a design which is based on reliability in fulfilling mission requirements and the use of state-of-the-art materials.

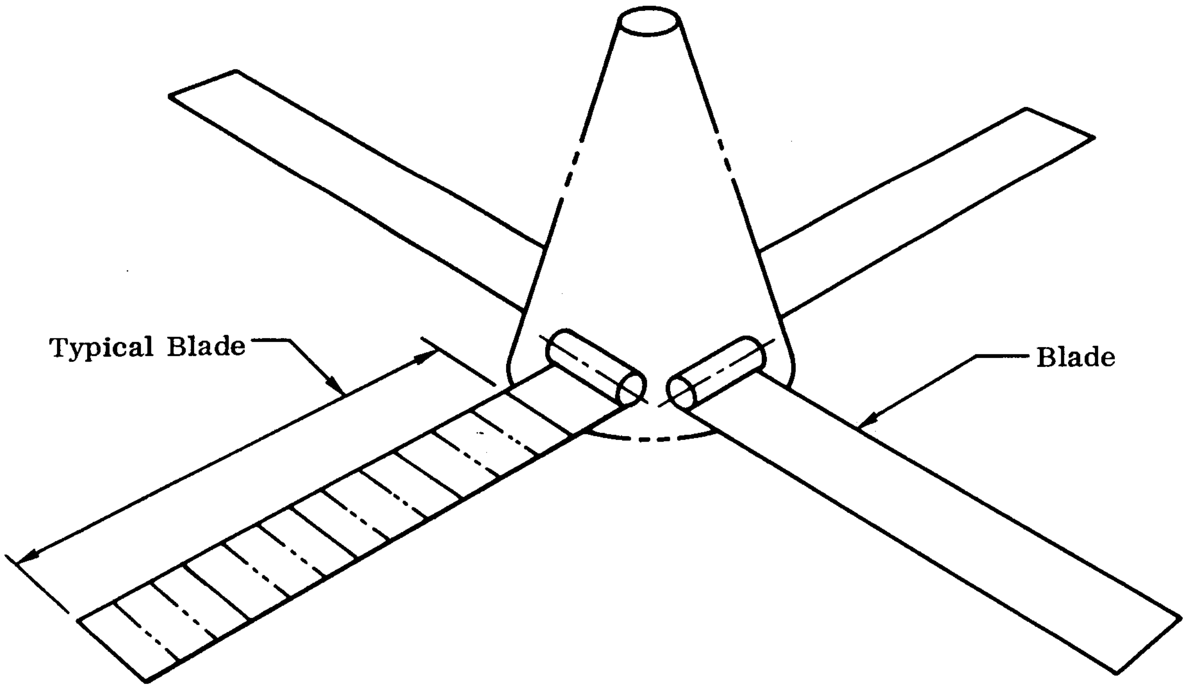

Figure 1. Roll-Up Solar Array Concept

STRUCTURAL AND THERMAL CHARACTERISTICS

	Contract Requirements	Design	Actual
Solar Cell Area/Panel	Approx. 50 ft^2	51 ft^2	51 ft^2
Structure Weight per Unit Area of Solar Cells	<.3 lbs/ft^2	.435 to .477 lbs/ft^2	.497 lb/ft^2
Static Requirements			
Substrate Radius of Curvature	6 in. (min)	6 in.	6 in.
Dynamic Requirements			
Angular Change of Solar Cell Substrate/ Unit Length	<1 Degree/in.	.3 Degree/in.	Not Tested
Variation in First Mass Moment of Array and Support Structure for Stowed Position	<±5%	± 5.1%	Not Tested
Undamped First Cantilever Frequency of Deployed Array Structure	Between 0.5 & 5.0 cps	0.55 cps	Not Tested
Variation if Ratio of Stiffness to Mass of Deployed Array Structure	<±10%	±4.3%	Not Tested
Thermal Requirements			
Resistance to Sterilization	295° F	300° F	Not Tested
Emissivity of Rear Surface of Substrate	>.8	.90	

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3.0 DESIGN CRITERIA

The design criteria applicable to the Deployable Large Area Solar Array Structure are taken in whole from the requirements presented in the Contract Statement of Work.

3.1 CONFIGURATION DESIGN CRITERIA

No provisions shall be required on the array structure to support unrelated spacecraft components such as power regulating zener diodes, cold gas attitude control systems and/or vernier solar pressure vanes.

During the launch phase, the array structure shall remain within the envelope shown on JPL Drawing No. J-4190680 (Sheet 1). This drawing reflects the available packaging regions for a broad range of typical Mariner spacecraft systems under study for use in the 1969 - 197X era. The drawing depicts a standard Surveyor class shroud on an Atlas-Centaur vehicle. The spacecraft is arbitrarily defined to be an octagonal frame, fifty-eight inches across on the major diagonal. Primary array structure attachment to the spacecraft may be accomplished along any of the corners or vertical edges of the hypothetical spacecraft frame.

The basic array structure shall be designed to have a minimum number of different components. This requires that the total array structure be composed of not more than four sub-elements or panels.

Total available surface for solar cell mounting shall be between one hundred fifty and four hundred square feet. For initial planning and conceptual study purposes, a target area of two hundred square feet shall be assumed. The geometry of the array structure shall be based upon a rectangular modular solar cell array of 18 inches x 34 inches, having a weight of 0.30 pound per square foot. This weight shall include cells, filters, modular wiring, and secondary cabling.

Any mechanical latch points or devices located on the cell surface shall not shade the solar cell surface when the array structure is oriented ±5° from the normal incidence angle of illumination.

3.2 STRUCTURAL DESIGN CRITERIA

Under the hypothetical environmental conditions set forth in the Environmental Criteria for a useful life of eighteen months:

a. The array structure shall have the capability of surviving normal ground handling during fabrication, assembly, qualification testing and storage. The array structure shall also have the capability of being repaired when subject to minor damage.

b. The array structure shall have the capability of surviving all dynamic loads, including transportation, cruiser course correction, and retromaneuver at planetary encounter. It is implicitly assumed that the array structure will be in the undeployed configuration during launch and in the deployed configuration during course correction motions. Depending upon the nature of the array structure (deployed or undeployed), either configuration may or may not be used during the retromaneuver. The retromaneuver thrust shall not be used to initiate or power the retraction, if required, of the array structure. Upon the completion of the retromaneuver, the array structure shall be in the deployed configuration suitable for power production.

c. The rear surface of the array structure shall be designed to minimize heat radiation traps in order to minimize local front surface hot spots.

d. All array structure components shall have provisions for pressure equalization between internal elements and the external flight environment.

e. To preclude real or potential degradation of the solar cells mounted upon the array structure, the curvatures induced in the cells shall be limited as follows:

   1. The radius of curvature of the undeployed or stowed array structure shall at no time be less than six inches.

   2. Under dynamic conditions, the angular change of the cell substrate per unit length shall be less than 1.0 degree/inch.

f. To avoid servoelastic coupling of the array structure and hypothetical spacecraft control system, the intertial and response characteristics for the array structure shall be as follows:

1. During powered flight (boost, retro), due to allowable tolerance variations in the fabrication process, the first mass moment of the array structure (undeployed or deployed) shall vary less than 5% as measured about the spacecraft centerline (boost axis).

2. If deployed, the array structure shall further have the following characteristics:

   a) The undamped first cantilever natural frequency of the array structure shall be between 0.5 and 5.0 cps.

   b) The ratio of damping to critical damping in the first cantilever mode shall be in the range 0.15 to 0.7.

   c) In the first cantilever mode of the array structure, the ratio of generalized stiffness to generalized mass (k/m) shall vary less than 10% due to all allowable tolerance variations in the fabrication processes.

3. During cruise phase (including course correction maneuver), the requirements shall be as defined in Paragraphs 3.2(f)2(a) and 3.2(f)2(c).

g. Structural criteria given below are stated in terms of limit loads (yield design loads). Induced stress levels shall be computed for all loading conditions stated in Paragraphs 3.2(g)1 and 3.2(g)2. Critical conditions shall be clearly identified and carefully evaluated. Margins of safety on stresses induced by these limit design loads as follows:

$$\text{M.S.} = \frac{\text{Y.S.}}{\text{L.S.}} - 1 > 0$$

$$\text{M.S.} = \frac{\text{U.S.}}{(1.25)(\text{L.S.})} - 1 > 0$$

where

M.S. = Margin of Safety
L.S. = Stresses resulting from Limit Loads
Y.S. = Yield Stress
U.S. = Ultimate Stress

The yield and ultimate stress values shall be those for the appropriate material as given in the latest editions of MIL-HDBK-5 and MIL-HDBK-17.

1. Thermal Cycling: this cycling represents the effects of spacecraft orbit about a planet as well as spacecraft attitude reorientations associated with mid-course corrections. The design limit thermal loads for this array structure are equivalent to the levels experienced during the following test environment:

   a) Pressure - The maximum pressure shall be 10^{-4} mm Hg.

   b) Free space background - The free space background or heat sink shall be simulated by a blackened wall having a total absorptivity or greater than 0.80 at liquid nitrogen temperatures, as viewed from the array structure.

   c) Heat Cycling - A heat input to the array structure surface of 80 watts per square foot shall be held until temperature stabilize. The electrical power source shall then be turned off for a 1-1/2 hour period. The subsequent step changes in electrical power input from 0 to 80 watts per square foot defines the start of a cycle. Periods of applied electrical power shall be for a 1-1/2 hour duration. Periods of nonapplied electrical power shall be for a 1-1/2 hour duration (see Figure 2). The array structure shall be subjected to 10 periods of applied heater power for a total test time of approximately 40 hours.

2. Limit Structural Design Loads: the following table contains the applicable limit accelerations for use in the determination of the appropriate limit loads. These accelerations define the environment at the array structure-spacecraft interface. The array structure shall be checked for structural adequacy under both static and vibratory criteria. Static and vibratory loads are not to be superimposed for design purposes.

Condition	Static Long	Static Lat.
(a) Max q and Mach 1	4g	3g
(b) Booster Burnout	12g	2g
(c) Booster Tailoff	2g	0
(d) Cruise Maneuver	0.2g	0.05g
(e) Retro Burner	6g	1g

	Vibratory		
Condition	Level	Range (cps)	Rate Min/Octave
(a) Max q and Mach 1	-	-	-
(b) Booster Burnout	1.6g rms	2 - 20	1 min/oct
	4.0g rms	20 - 200	1 min/oct
	Noise $0.2g^2$/cps	200-2000	180 seconds
(c) Booster Tailoff	-	-	-
(d) Cruise Maneuver	0	-	-
(e) Retro Burner	8g rms	2-20	0.5 min/oct
	2.0g rms	20-200	0.5 min/oct
	Noise $0.2g^2$/cps	200-2000	180 seconds

3.3 THERMAL DESIGN CRITERIA

All components shall meet the following sterilization requirements:

a. Withstand exposure to three 36-hour periods of heat at 145° C (295° F) in dry nitrogen (a total of one hundred eight (108) hours).

b. Withstand exposure to a gas mixture of 12% ethylene oxide, 88% freon gas for ten hours at a relative humidity between 30% and 50%.

The temperature at any point on the solar cell surface, as a function of solar irradiance, shall be less than the values defined in Figure 3. This is a maximum temperature for any position on the front surface of the array structure. These temperatures may be achieved by requiring that exposed surfaces on the rear and edges of the array structure have a total hemispherical emissivity of greater than 0.80 in the temperature range of -30° C and 80° C.

The rear surface of the array structure shall be designed to minimize heat radiation traps in order to minimize local front surface "hot spots".

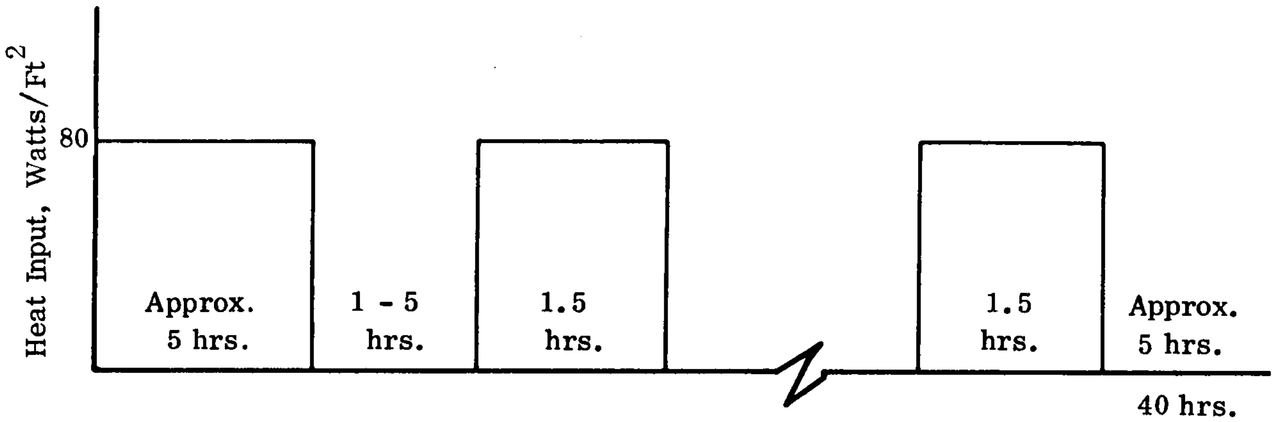


Figure 2. Periods of Electrical Power

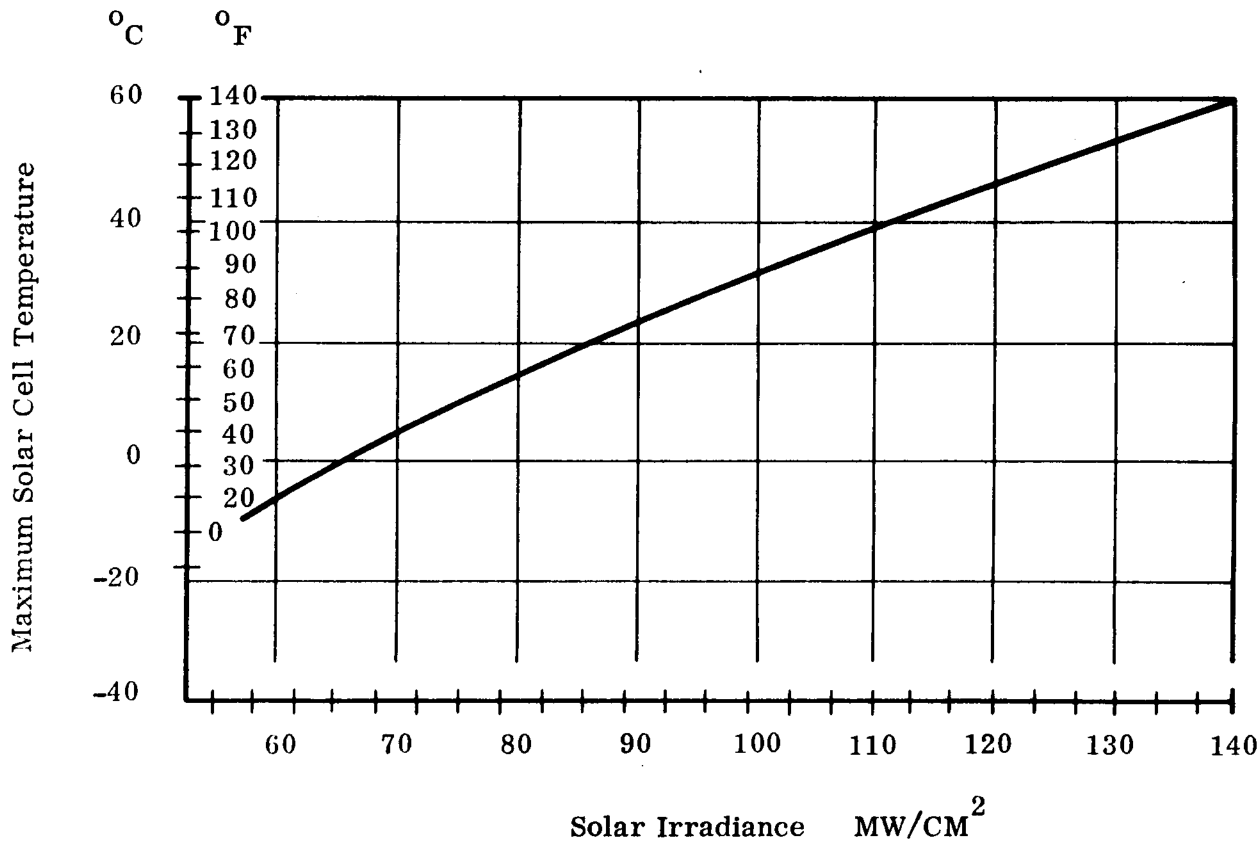


Figure 3. Solar Irradiance Versus Maximum Solar Cell Temperature

Thermal cyling represents the effect of spacecraft orbit about a planet as well as spacecraft attitude reorientations associated with cruise course corrections. The design limit thermal loads for this array structure are equivalent to the levels experienced during the following test environment:

a. Pressure - The maximum pressure shall be 10^{-4}mm Hg.

b. Free Space Background - The free space background or heat sink shall be simulated by a blackened wall having a total absorptivity of greater than 0.80 at liquid nitrogen temperatures as viewed from the array structure surface.

c. Heat Cycling - A heat input to the array structure surface of 80 watts per square foot shall be held until temperatures stabilize. The electrical power source shall then be turned off for a 1-1/2 hour period. The subsequent step changes in watts/square foot define the start of a cycle. Periods of applied electrical power shall be for a 1-1/2 hour duration. Periods of nonapplied electrical power shall be for a 1-1/2 hour duration. The array structure shall be subjected to ten periods of applied heater power for a total test time of approximately forty hours.

3.4 MATERIAL DESIGN CRITERIA

The cell mounting surface shall be capable of being cleaned with solvents or mild acid etching techniques prior to cell mounting.

The cell mounting surface shall be fabricated of or coated with a material that is an electrical insulator. This material shall be capable of withstanding the rigors of cell-mounting techniques. This material shall survive and be capable of repair, in the event that a damaged or defective cell must be removed. The insulation resistance shall be greater than 100 megohms, measured at a test potential of 200 VDC between the cell mounting surface and any metallic portion of the substrate.

The use of any material shall be predicated upon the proven ability of the material to withstand the deep space environment for a time in excess of eighteen months.

All components shall meet the following sterilization requirements:

a. Withstand exposure to three thirty-six hour periods of heat at 145° C (295° F) in dry nitrogen, a total of one hundred eight (108) hours.

b. Withstand exposure to a gas mixture of 12% ethylene oxide, 88% freon gas for ten hours at a relative humidity between 30% and 50%.

The exposed surfaces on the rear and edges of the array structure shall have a total hemispherical emissivity of greater than 0.80 in the temperature range of -30° to 80° C.

Magnetic materials shall not be used in any of the array structure components, except when array structure reliability is affected by use of such materials.

3.5 WEIGHT DESIGN CRITERIA

A design objective shall be to keep the weight of the array structure and deployment mechanisms below 0.6 pound per square foot, including solar cells, cabling and wiring.

The solar cell array shall have a weight of 0.30 pound per square foot. The weight shall include cells, filters, modular wiring and secondary cabling.

4.0 DESIGN DISCUSSION

4.1 DESIGN CONCEPT

Past experience with rigid solar panel substrate structures has shown that minimizing panel deflection due to dynamic loads becomes an increasingly difficult design task as panel size increases. This task could be more easily accomplished if allowable weights could be increased to provide stiffer substrate structures or adequare damping devices. The demand for advancing the state-of-the-art, however, requires that allowable weights be reduced rather than increased. This fact causes a basic incompatibility in the concept of a large rigid array that is also light in weight.

The roller drum concept reduces the dynamic deflection problems by providing a highly rigid shape due to its large diameter and short length. Using this rigid structure to support a damped flexible substrate, maximum advantage of the rigid structure may be realized during the periods of high loads. The design also uses a relatively lightweight substrate and beam assembly to react the greatly reduced g loads experienced after deployment.

Although the roller drum concept is now new, it has not been utilized to date. Two major reasons for this are: (1) the requirement for large arrays was not critical, (2) a reasonable method of extending the supporting substrate was not available.

The development of a lighweight retracting and extending beam compatible with a roller is the major reason for the feasibility of the concept. The first phase effort developed a beam that fulfills these goals and thus becomes a roller concept that is logical and workable for packaging and extending large area solar cell arrays.

The array concept, shown schematically in Figure 4, and in detail by Figures 5 and 6, is designed to deploy a minimum of 200 foot^2 of solar cell area. This area is divided into four blades of approximately 50 square feet each. These blades are 38.5 inches x 18.6 feet.

Each blade contains four removable substrate modules. The particular design presented fits well within the envelope of a hypothetical spacecraft.

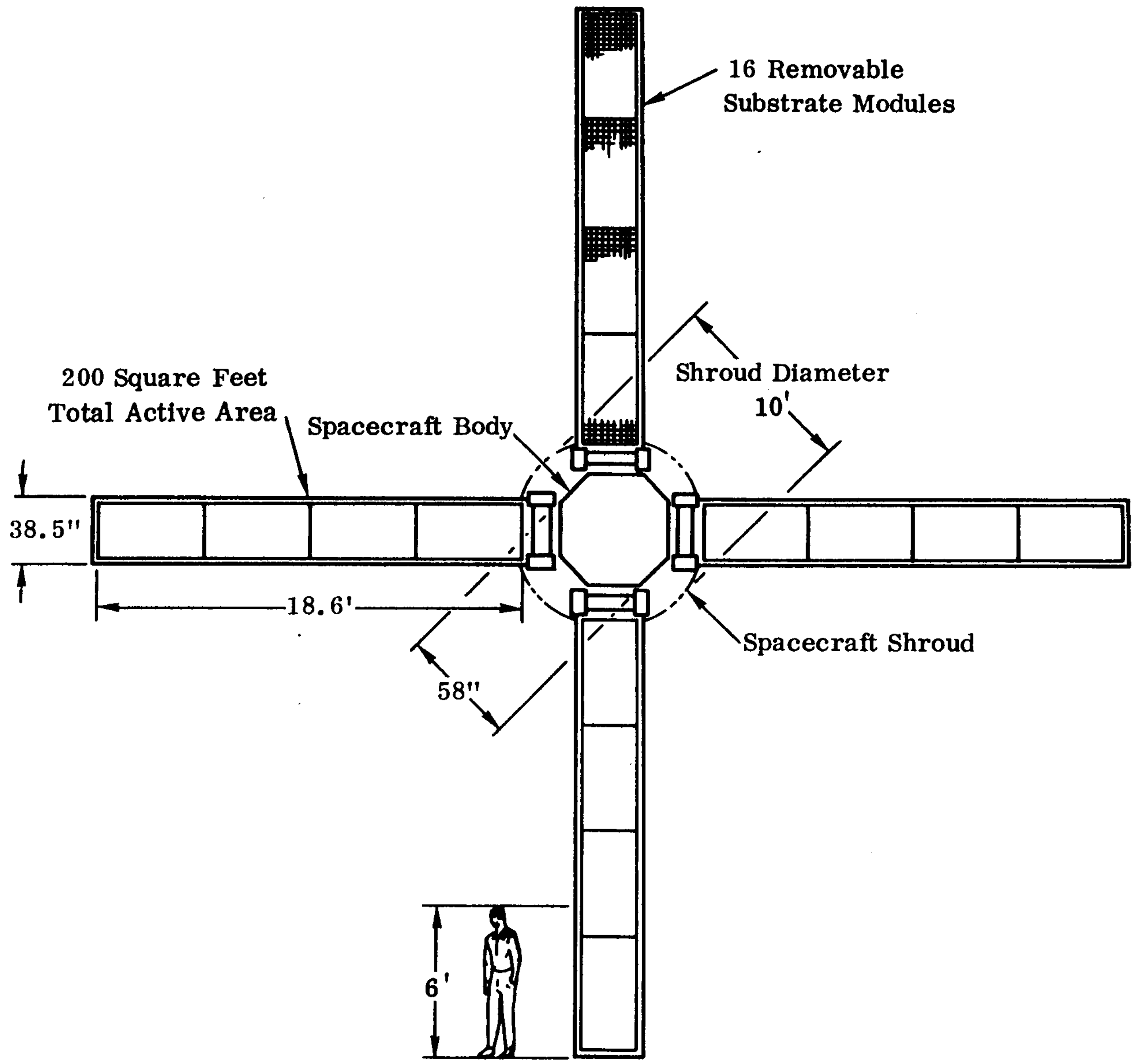
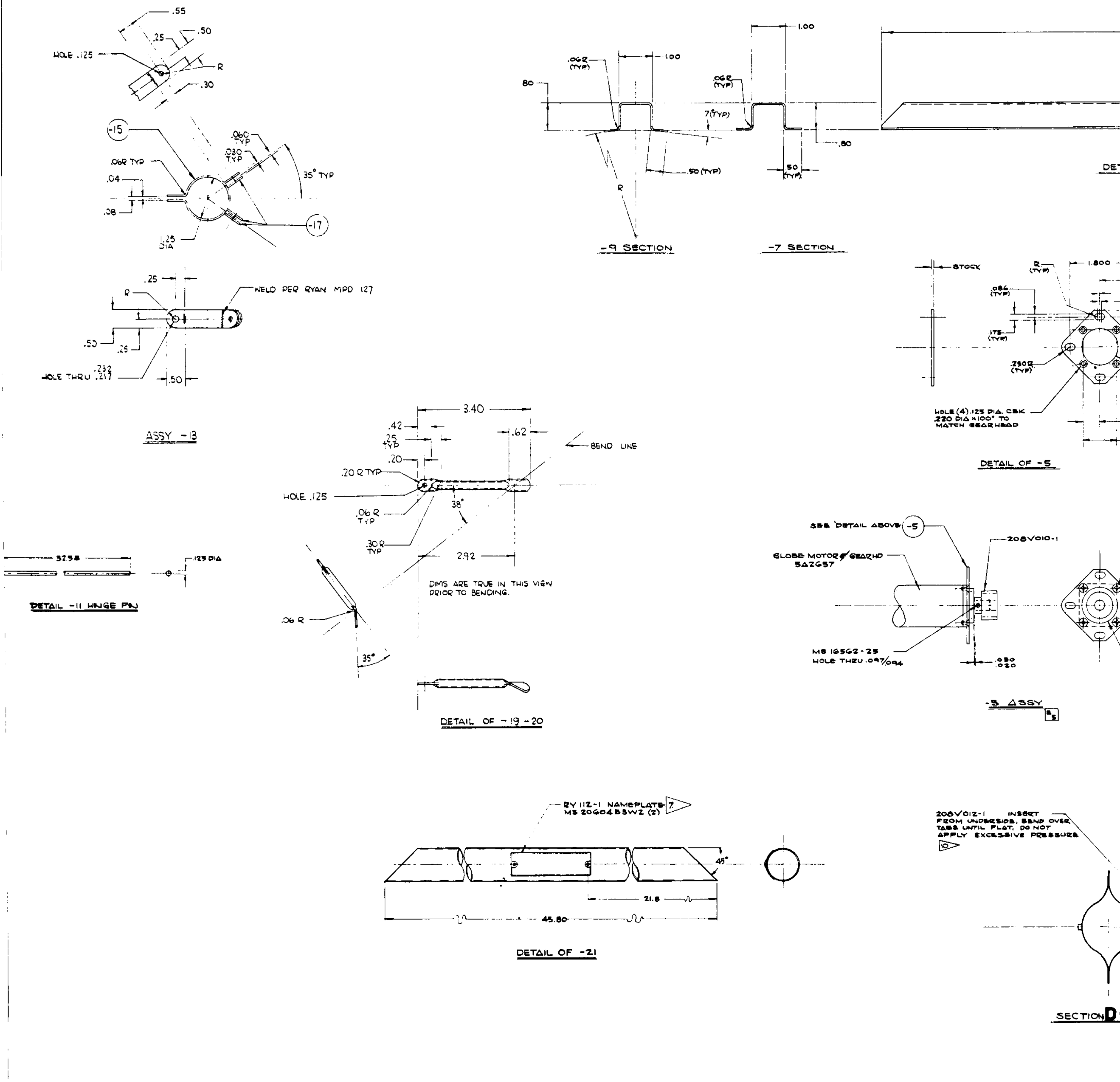
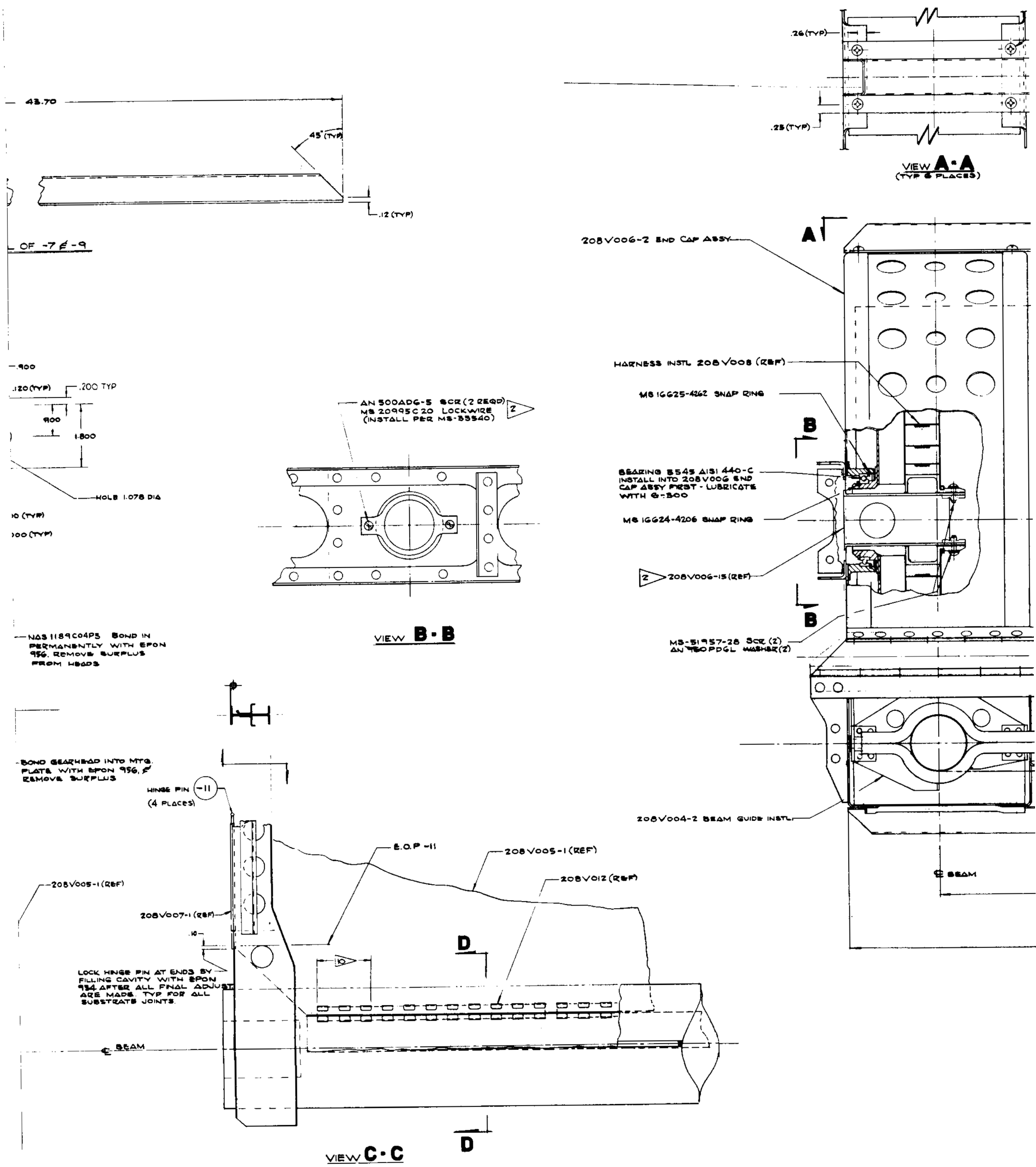
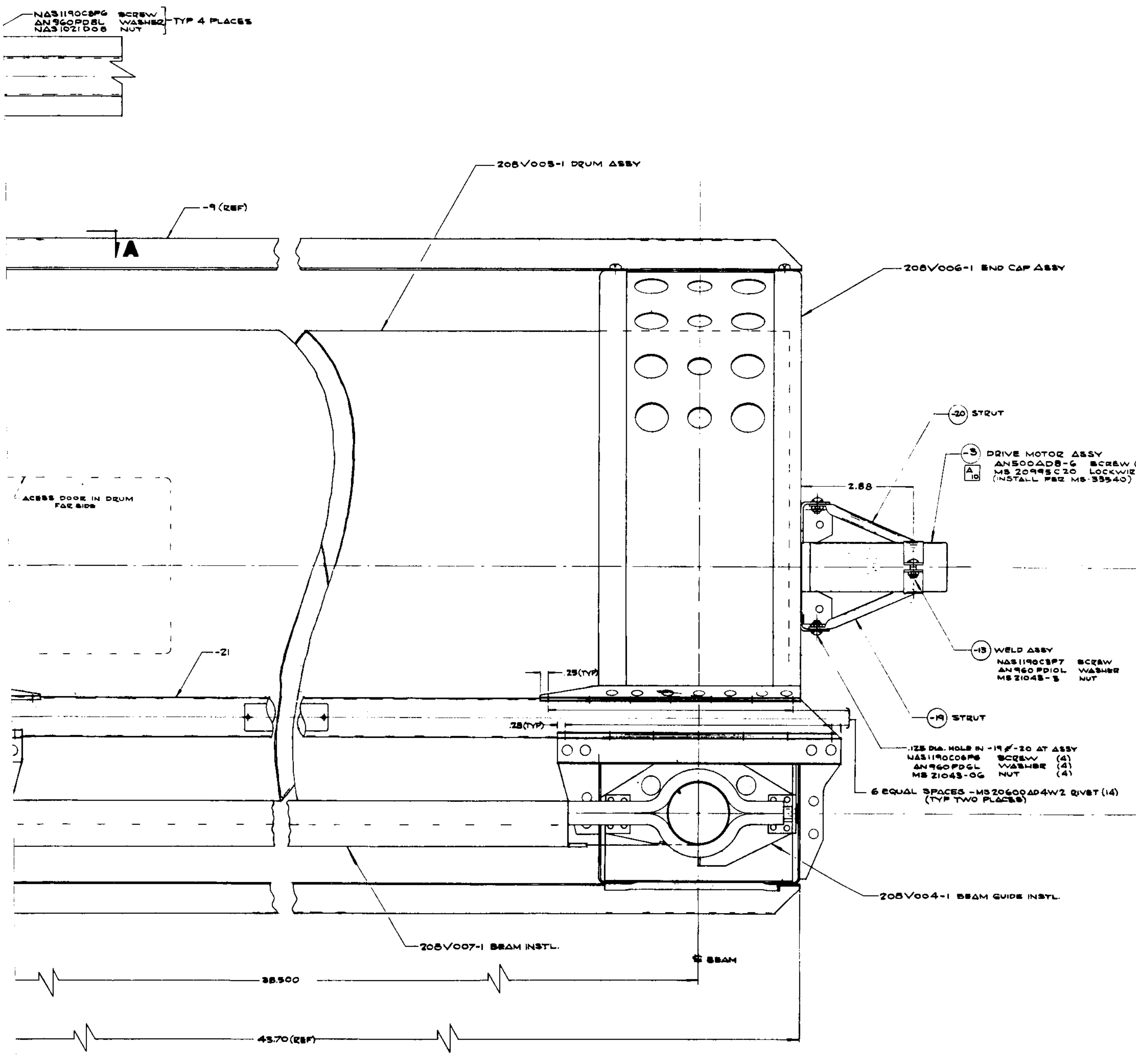


Figure 4. Solar Array Plan Schematic for the Hypothetical Spacecraft

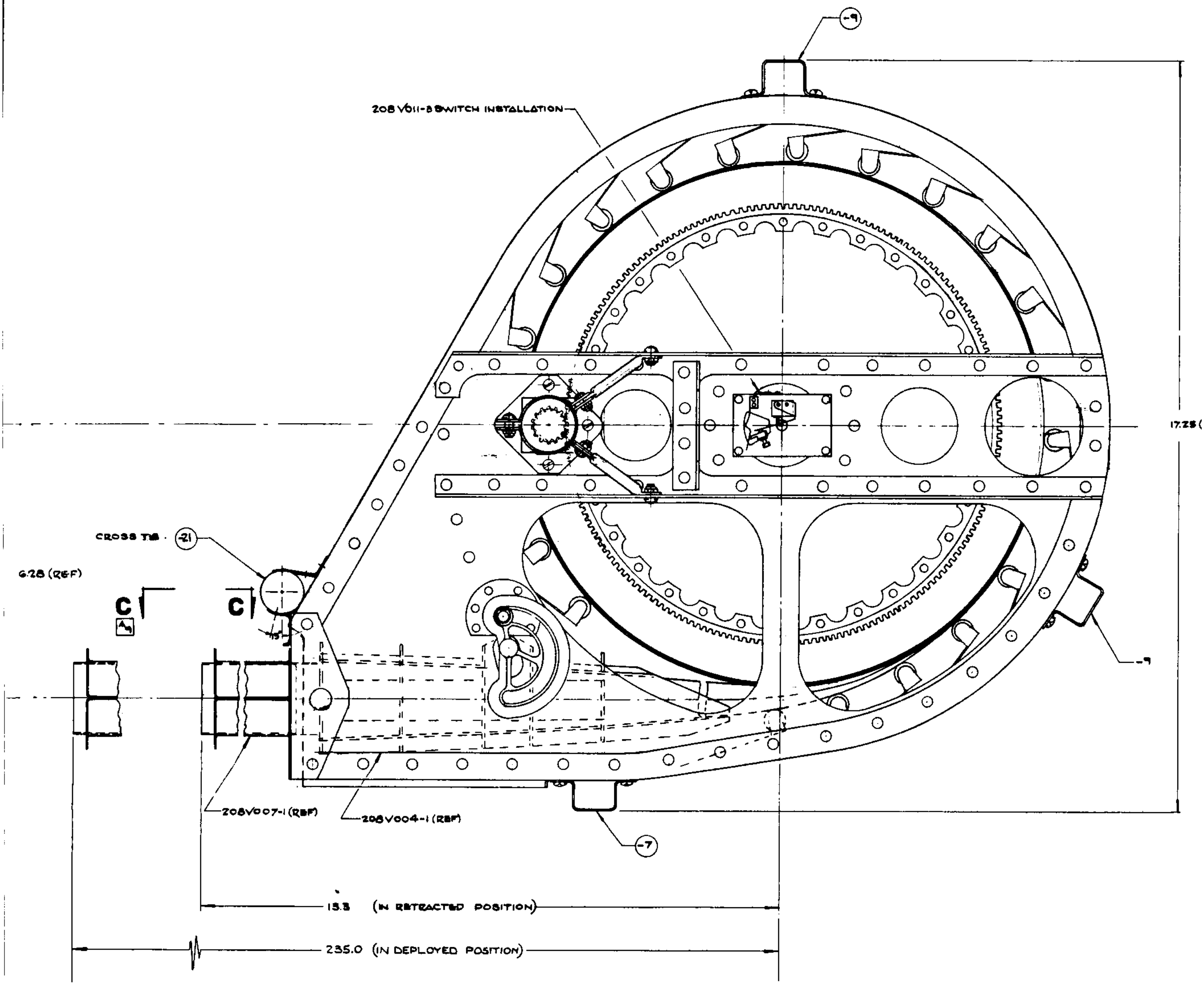

HOLE .125
.55
.25
.50
R
.30
-15
.060 TYP
.030 TYP
.06R TYP
.04
.08
35° TYP
-17
1.25 DIA
.25
R
WELD PER RYAN MPD 127
.50
.25
HOLE THRU .232 .217
.50
ASSY -13
3.40
.42
.25 TYP
.62
BEND LINE
.20
.20 R TYP
HOLE .125
.06 R TYP
38°
.30 R TYP
2.92
DIMS ARE TRUE IN THIS VIEW PRIOR TO BENDING.
.06 R
35°
DETAIL OF -19 -20
DETAIL -11 HINGE PIN
.125 DIA
.06R (TYP)
1.00
80
R
.50 (TYP)
-9 SECTION
.06R (TYP)
7°(TYP)
.80
.50 (TYP)
-7 SECTION
STOCK
R (TYP)
1.800
.086 (TYP)
.175 (TYP)
.250R (TYP)
HOLE (4) .125 DIA. CSK .220 DIA x100° TO MATCH GEARHEAD
DETAIL OF -5
SEE DETAIL ABOVE -5
208V010-1
GLOBE MOTOR & GEARHD 5A2657
MS 16562-25 HOLE THRU .097/.094
.030 .020
-3 ASSY
RY 112-1 NAMEPLATE 7
MS 20604 B3W2 (2)
45°
21.8
45.80
DETAIL OF -21
208V012-1 INSERT FROM UNDERSIDE, BEND OVER TABS UNTIL FLAT. DO NOT APPLY EXCESSIVE PRESSURE
10
SECTION D



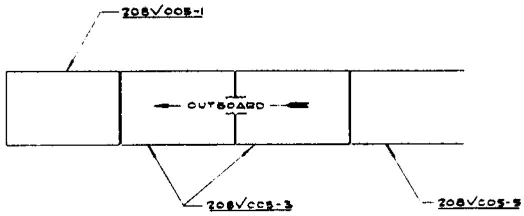
43.70
45° (TYP)
.12 (TYP)
OF -7 & -9
.26 (TYP)
.25 (TYP)
VIEW A-A
(TYP 6 PLACES)
208V006-2 END CAP ASSY
A
HARNESS INSTL 208V008 (REF)
MS 16625-4262 SNAP RING
B
BEARING B545 AISI 440-C INSTALL INTO 208V006 END CAP ASSY FIRST - LUBRICATE WITH G-300
MS 16624-4206 SNAP RING
208V006-15 (REF)
MS-51957-28 SCR (2)
AN 960PDGL WASHER (2)
208V004-2 BEAM GUIDE INSTL
℄ BEAM
.900
.120 (TYP)
.200 TYP
.900
1.800
HOLE 1.078 DIA
AN 500AD6-5 SCR (2 REQD)
MS 20995C20 LOCKWIRE
(INSTALL PER MS-33540)
2
VIEW B-B
NAS 1189C04P5 BOND IN PERMANENTLY WITH EPON 956. REMOVE SURPLUS FROM HEADS
BOND GEARHEAD INTO MTG. PLATE WITH EPON 956, REMOVE SURPLUS
HINGE PIN -11
(4 PLACES)
208V005-1 (REF)
208V007-1 (REF)
.10
LOCK HINGE PIN AT ENDS BY FILLING CAVITY WITH EPON 934 AFTER ALL FINAL ADJUST. ARE MADE. TYP FOR ALL SUBSTRATE JOINTS.
E.O.P -11
208V005-1 (REF)
208V012 (REF)
10
D
℄ BEAM
VIEW C-C



(-1) DEPLOYABLE SOLAR PANEL ASSEMBLY



DRUM ASSY SHOWN IN DEPLOYED POSITION

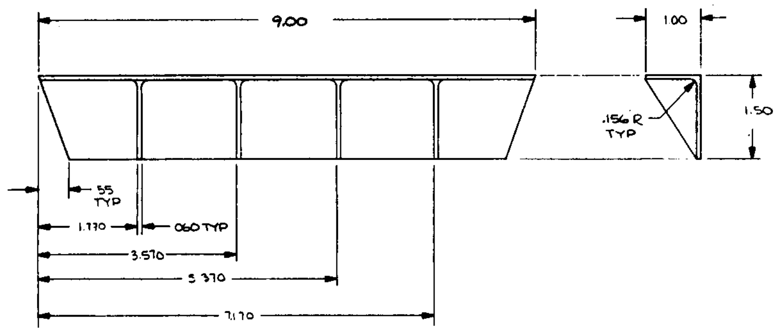


VIEW SHOWING RELATIVE POSITIONS OF SUBSTRATE ASSEMBLIES IN DEPLOYED CONDITION

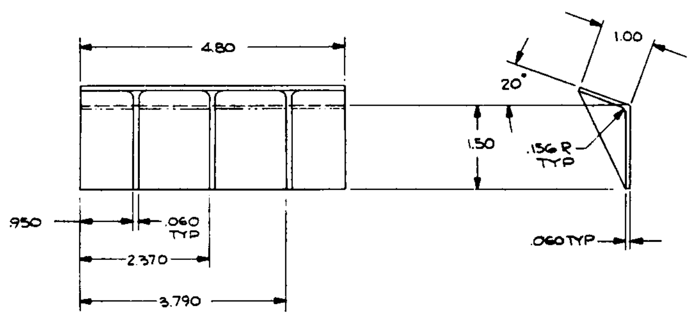
NOTES;

1. MAY BE PURCHASED FROM GLOBE INDUSTRIES-DAYTON 4, OHIO
2. SEE 208V008 FOR INSTALLATION INSTRUCTIONS.
3. ALL PROCESSES TO BE IN ACCORD WITH RYAN SPEC. 208S002
4. HEAT TREAT TO T-6 AFTER FORMING
5. ANODIZE WITH DOW 17 TYPE I PER MPD-102
6. SHELL CHEMICAL AND PAINT CO., PITTSBURG, CALIF.
7. ADD THE FOLLOWING INFORMATION TO NAMEPLATE:
   NAME OF PART: DEPLOYABLE SOLAR PANEL
   SERIAL NO.: 1, 2, 3, 4 (AS APPLICABLE)
   PART NO.: 208V001
   CONTRACT NO.: NAS 7-100/951107
   MARK OUT "FSN"
8. THIS PART TO BE VERIFIED FOR AVAILABILITY. INFORMATION TO BE SUPPLIED AT A LATER DATE.
9. CHEM MILL WALL THICKNESS TO .042 -.044 PER MPD 122. CHEM MILL O/D OF TUBE ONLY.
10. 208V0124 INSERTS TO BE INSTALLED IN SUCESSION WITH THE LAST ONE TO BE TRIMMED TO FIT REMAINING HOLES BETWEEN SUBSTRATE & BEAM. ONE (1) INSERT REQUIRED TO TRIM EACH SUBSTRATE SECTION.

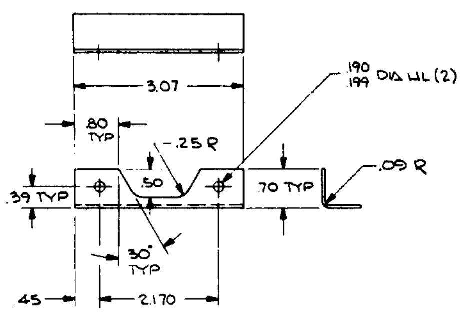
Figure 5. Deployable Panel Assembly



DETAIL -15 B



DETAIL -17 B



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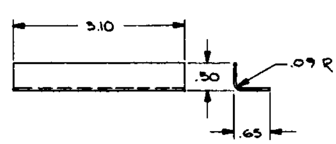
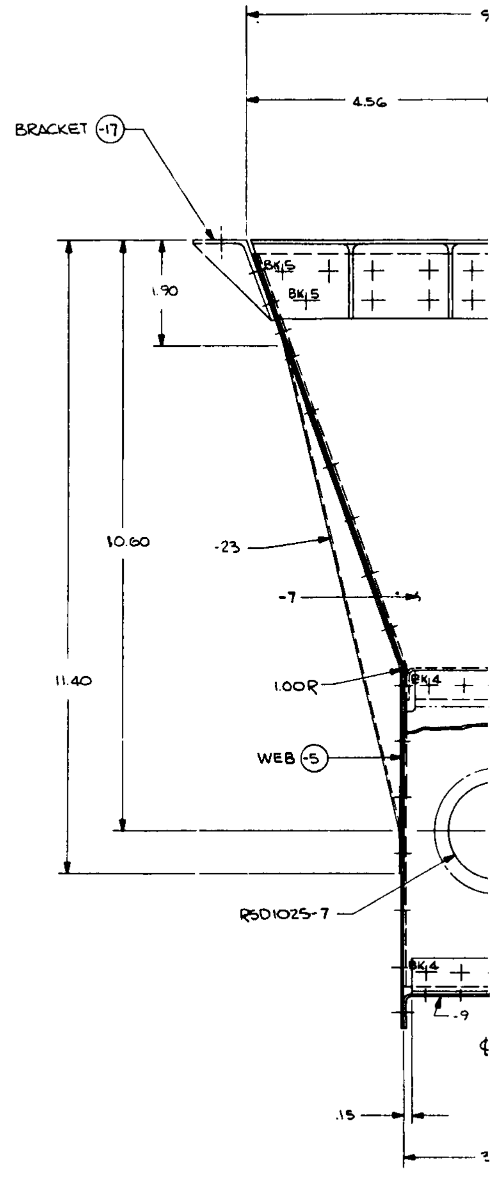
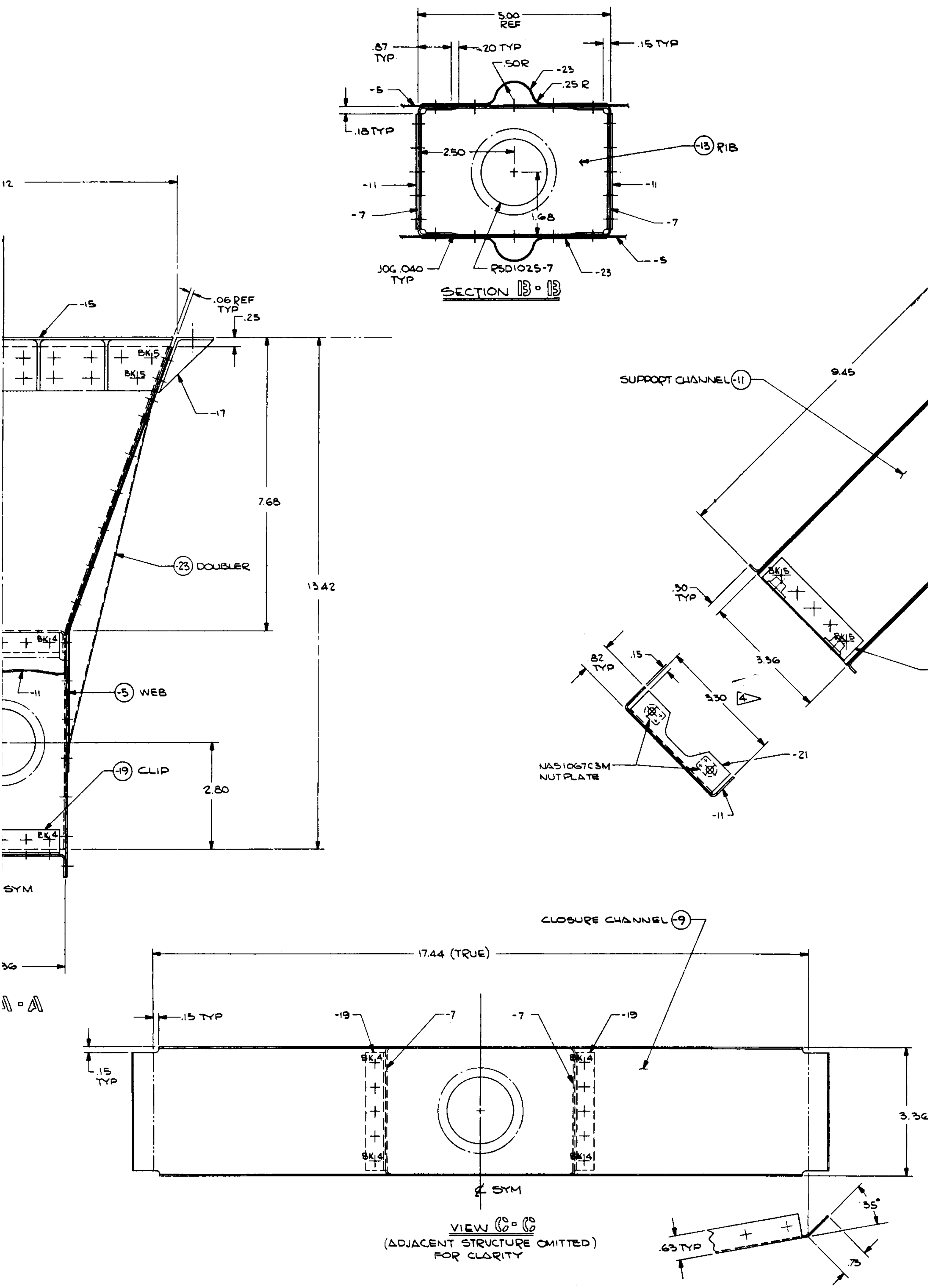


DETAIL -19



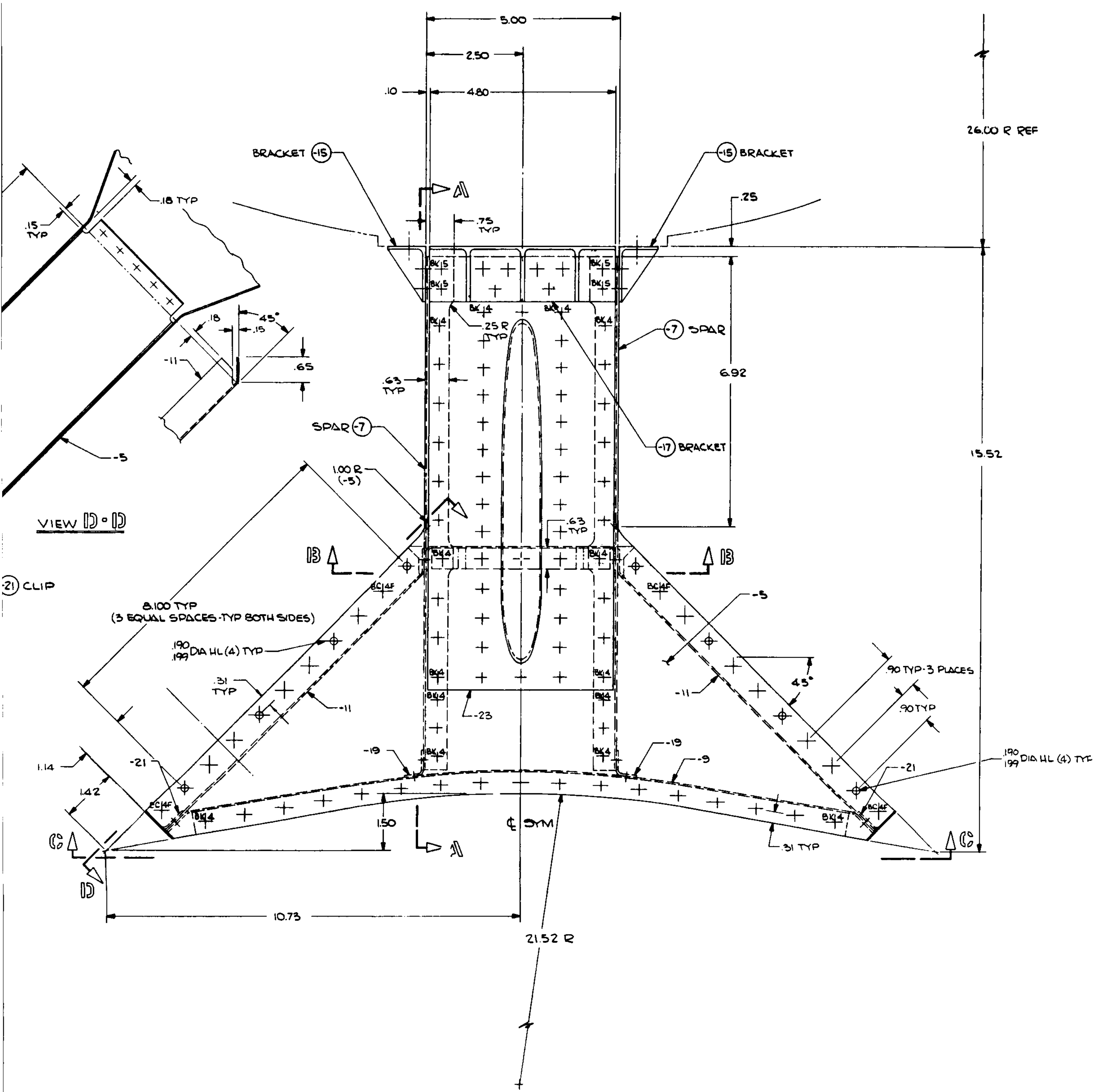
SECTION



SECTION B-B

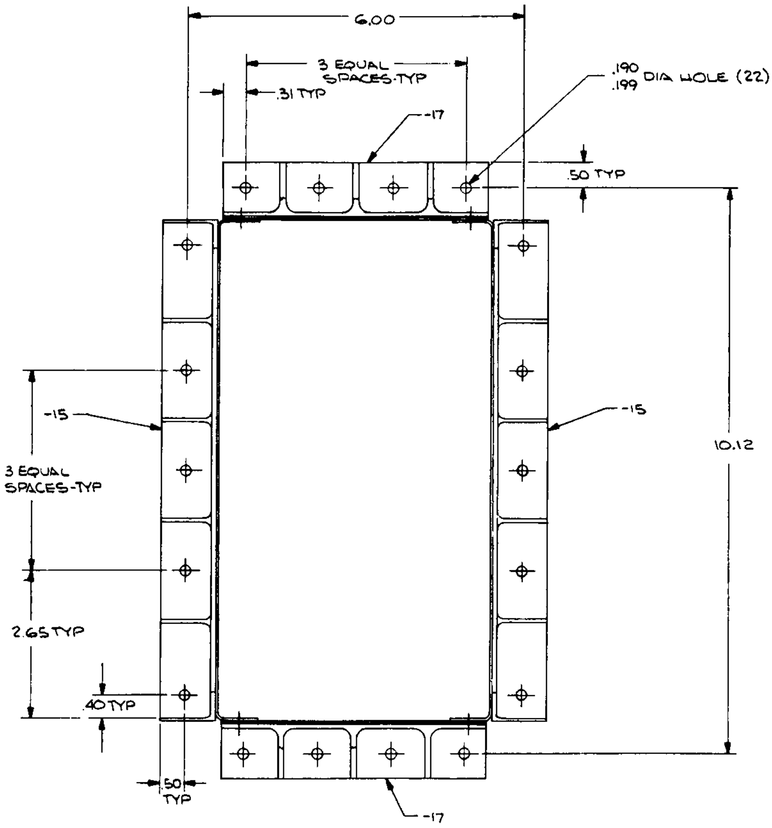
VIEW C-C
(ADJACENT STRUCTURE OMITTED)
FOR CLARITY

A-A

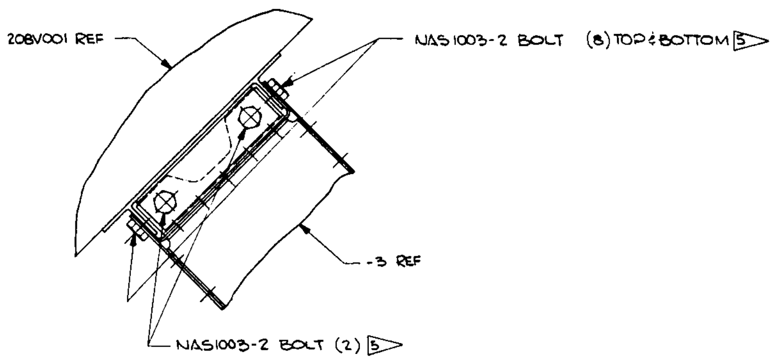


PLAN VIEW

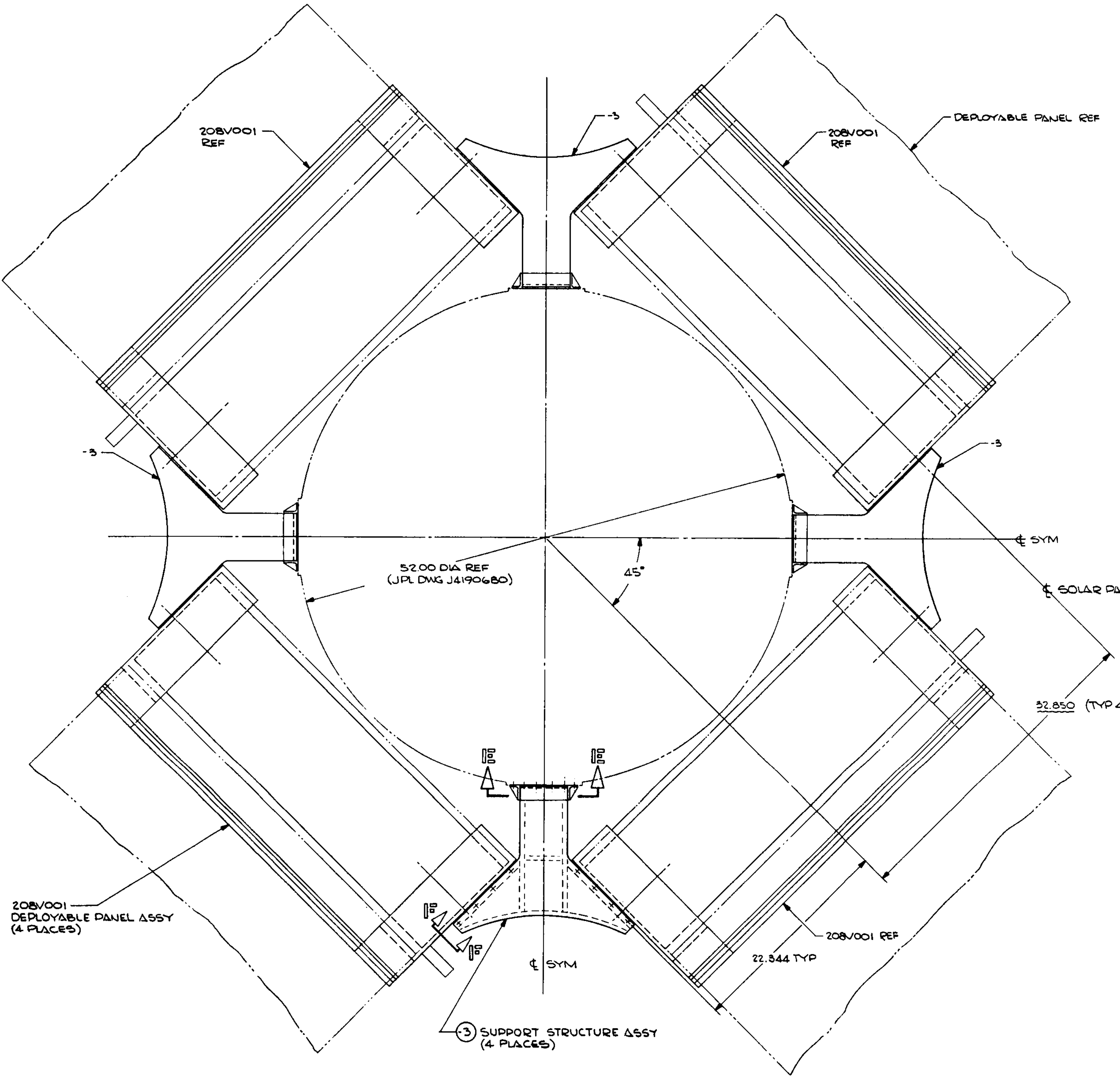
-3 SUPPORT STRUCTURE ASSEMBLY



SECTION E - E
(SHOWING ATTACH PATTERN -1 INSTL TO VEHICLE-TYP 4 PLACES)



SECTION F - F
(TYP 8 PLACES)



-1 SOLAR ARRAY INSTL

SCALE 1/4

NOTES:

1. EDGE DISTANCE FOR NO. 4 RIVETS TO BE .25 EXCEPT AS SHOWN.
2. EDGE DISTANCE FOR NO. 5 RIVETS TO BE .31 EXCEPT AS SHOWN.
3. ATTACHMENT HARDWARE FOR -1 INSTL TO BE FURNISHED BY CUSTOMER.
4. MATING DIMENSION $^{+.01}_{-.00}$. TO ALLOW FITNESS OVER ATTACH FLANGES OF 208V001.
5. TORQUE TO 20-25 IN. LBS.
6. BEND RADII TO BE .09.
7. RELIEF RADII TO BE .12.
8. ALLOWABLE MACHINE MISMATCH FROM DIMENSIONS INDICATED TO BE -.000 TO +.030.
9. SHEET STOCK TOL $^{+.00}_{-STD}$.
10. ALL PROCESSES TO BE IN ACCORD WITH RYAN SPEC 2085002
11. ANODIZE ALL DETAILS WITH DOW 17 TYPE I PER MPD 102
12. REMOVE 208V001-7 & -9 CHANNELS ON INSTL.

Figure 6. Solar Array Installation

When completely retracted, the rectangular envelope of the structure is 17 inches by 21 inches by 49 inches. The total volume is 10 cubic feet. Actual space availability and interface structure can be altered to provide even greater capability.

An electrical layout was made using the available area and geometry of the structure, see Figure 7.

Each blade contains four 55.8 inch x 36.64 inch modular panels. Each panel carries three 18 inch x 34 inch rectangular solar cell modules. This array is capable of producing 10 watts per square foot or 2,000 watts total power. An area 0.6 inch wide is furnished along the length of the substrate for routing flat electrical leads.

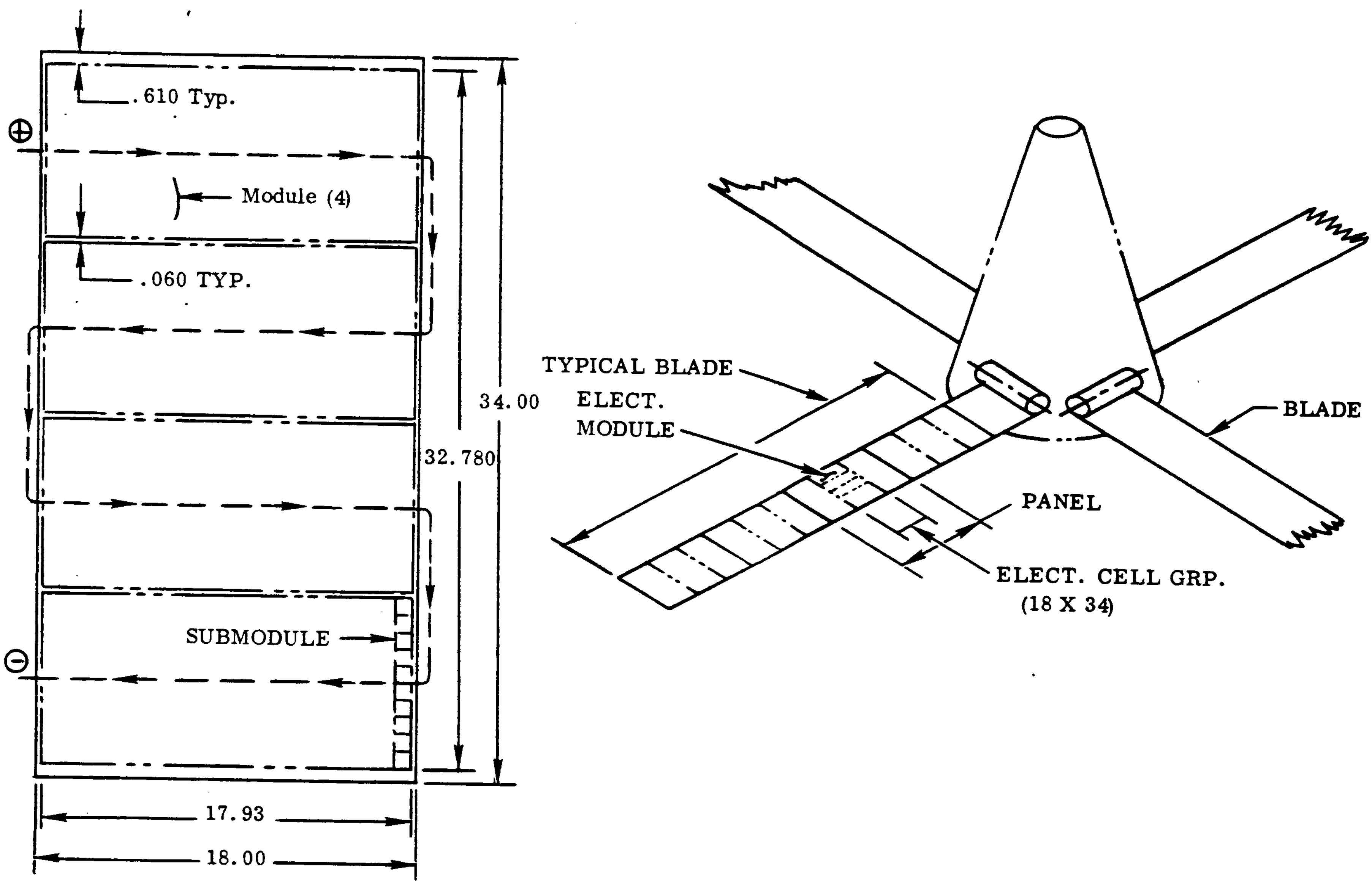
Figure 8 is an illustration of the solar array structure.

The panels are deployed and supported by extendable beams. The beams and solar cells substrate are stowed on a cylindrical drum which is driven by an electric motor to deploy or retract the solar panels. The support beams are designed to flatten as they approach the drum thus making wrapping possible.

4.2 SUPPORT BEAMS

The preliminary investigation conducted at the time of Ryan's proposal to JPL indicated the feasibility of rolling a structural beam around a one-foot diameter drum. Subsequent development of this basic idea during Phase I was aimed at optimizing the beam configuration to fulfill design parameters. The essential part of this development was the design of a shape which would be most effective to react the bending loads imposed by the 0.2g cruise maneuver. In addition, this shape should be one which would allow flattening of the section with relatively little load and no permanent distortion. Closely coupled to the design of the shape was the selection of a material which would tolerate flattening and wrapping without yielding, and one which would perform under all environmental conditions.

The beam shape presented in the proposal is shown as Design A, Figure 6. This shape was discarded after further analytical and sample test investigations. The shape selected is Design B of Figure 9.



(18 X 34) ELECTRICAL STRING

Figure 7. Solar Array

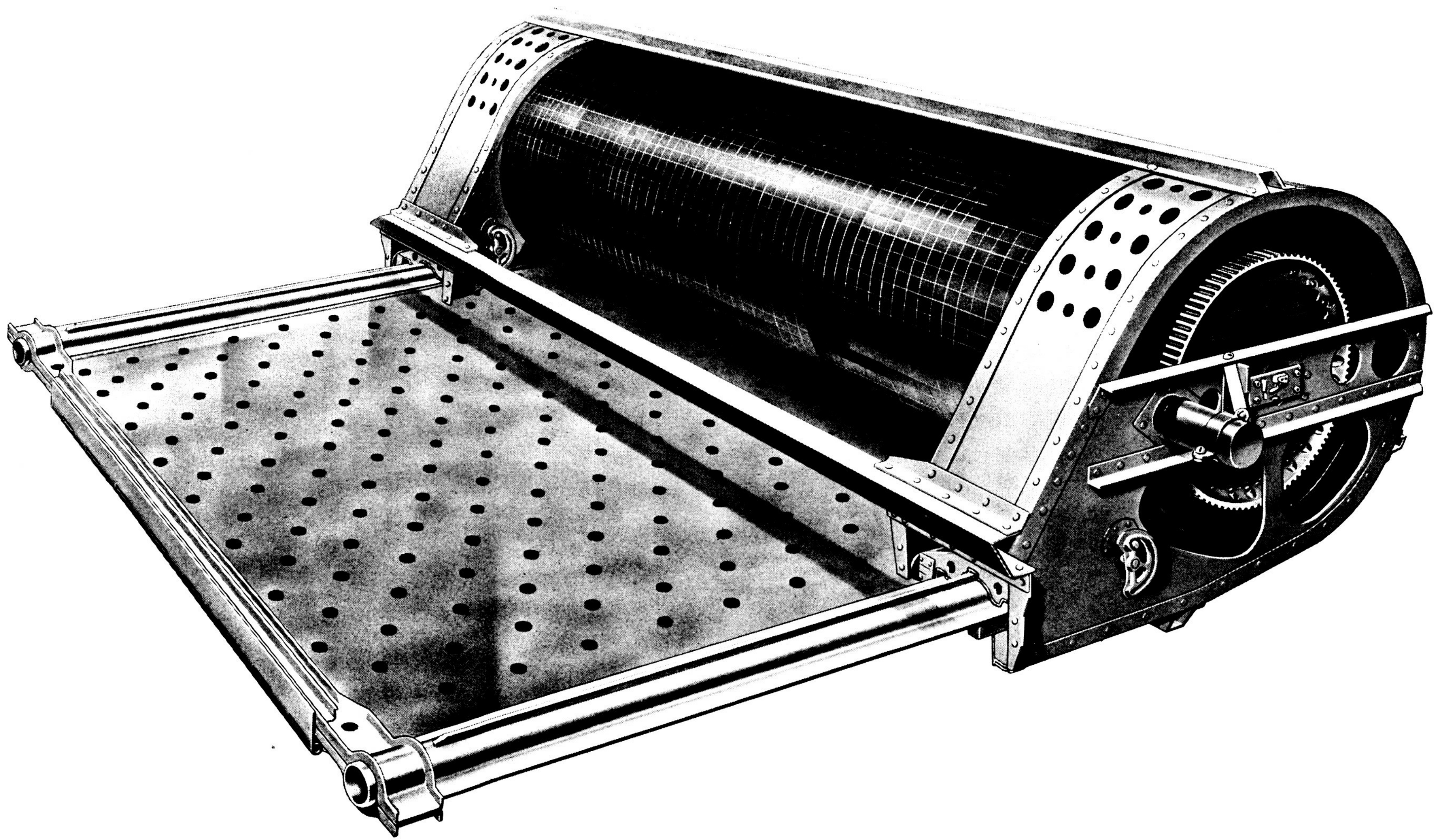
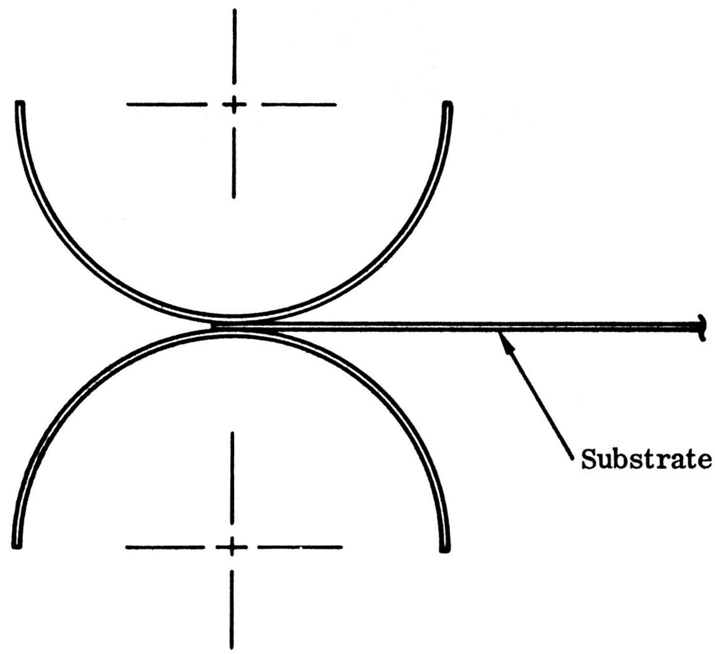
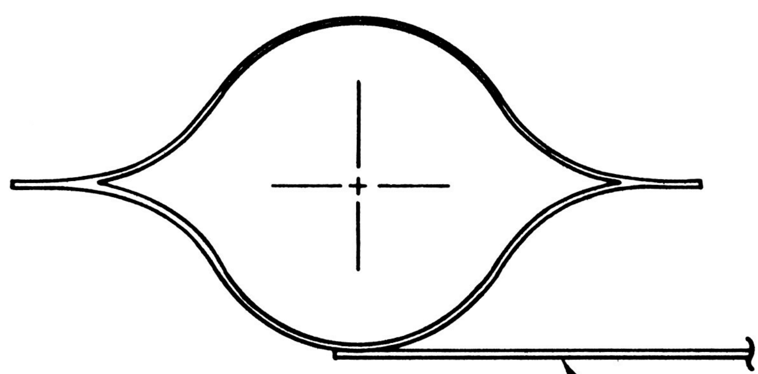


Figure 8. Deployable Array Structure



Design A
(Proposed)



Design B
(Selected)

Figure 9. Basic Beam Shapes

Reasons for the change to Design B were:

a. The unsupported edges of Design A proved to be critical in buckling when the load direction imposed compression loads on these edges. The side of the beam taking tension loads was required to support approximately 80% of the load. This characteristic caused the beam weight to be high in comparison to the load reacted, because only one-half of the beam assembly was working to react a load normal to the beam.

   Conversely, Design B eliminates the unsupported edges by creating a closed section stabilized for buckling by the reversed curvature of the shape. This design utilizes the full height of the beam by working both beam caps at one time, rather than one-half of the beam, to react loads from a particular direction.

b. The open section shown as Design A has very little torsional rigidity. Closing the section, as accomplished in Design B, greatly increases the torsional capability of the beams, thereby increasing the torsional rigidity of the entire panel.

Bending tests were conducted on various materials and section shapes of varying thicknesses. An optimum modulus to weight ratio was selected which was compatible with such other factors as weight, corrosion, magnetic aspects, creep, etc. This approach also reduced the cap stresses to an allowable value.

Material investigation was limited to those materials which were non-magnetic. Glass fiber reinforced plastic, aluminum, stainless steel, and titanium were used for the fabrication of test samples to check flattening characteristics. Aluminum and stainless steel were less efficient than titanium. The fiberglass thicknesses required necessitated excessive flattening loads, which rated this material as second choice. Problems of resin creep at temperature were overcome by the use of proper resins. Titanium exhibited all the desirable qualities needed, and test sample fabrication proved this material capable of fulfilling requirements.

The cross-sectional shape of the beam and substrate attachment is shown diagramatically in Figure 10. The substrate attach flange is seam-welded to the lower portion. Both sections are joined as indicated at their faying surfaces by seamwelds along the entire length. The inboard end of the beam assembly is preflattened to facilitate attachment to the drum. The outboard end of the beam attaches directly to a cross member leading to the beam on the opposite end of the panel assembly. The

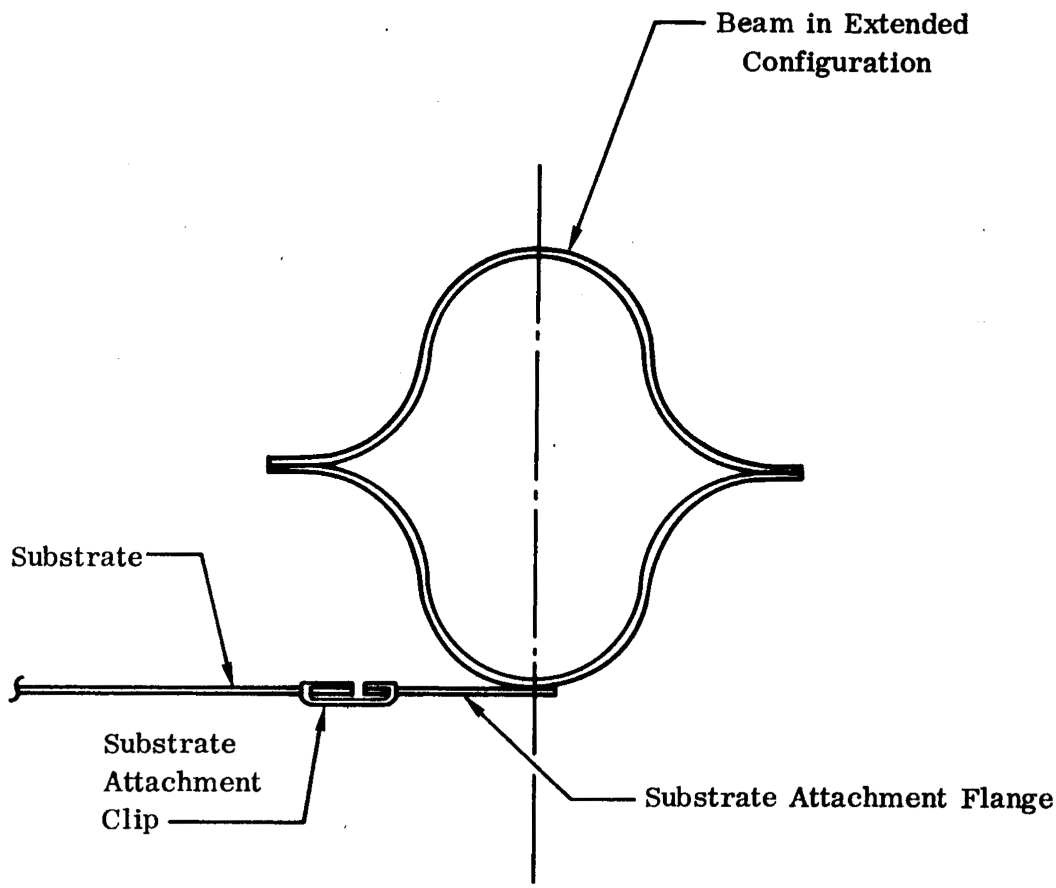


Figure 10. Section Through Deployable Beam

substrate attach flange contains precisely located slots which are matched to mating slots in the substrate and opposite hand beam. An aluminum substrate attachment clip is inserted into these slots to fasten the beams and substrate together.

4.3 SOLAR CELL ARRAY STRUCTURE

Figure 11 is a photograph of the solar cell array structure showing the rear surface of the beams and substrate. The roller drum mounted in bearings attaches to the beam and pulls the beam in for retraction or pushes the beam out for extension.

The beam is stowed by rotating the storage drum which reels in the beam, through a guide sleeve causing the beam to flatten against the drum. Twenty-two spring loaded rollers in each end assembly hold the beam against the drum. As the direction of drum rotation is reversed, the pressure on the beam is removed and the beam returns to its original shape. A silicone rubber spacer strip is bonded to the rear surface of the beam to provide a controlled space between the substrate layers as they wrap on to the storage drum. The drum is rotated on its bearings by the drive motor working through a spur gear.

The drive motor is a 24 volt dc unit with an integral planetary gear reduction cage. The motor speed is 12,000 rpm. The gear box output shaft speed is 18.75 rpm. The ratio of drive pinion gear diameter to spur gear diameter produces a roller drum speed of 1.41 rpm and a lineal beam extension rate of 4.43 feet/minute. The torque capability of the motor is one inch-ounce which produces 219 inch-pounds of torque at the roller drum. The power required for extension of the beam is 7.5 watts; for retraction 10.0 watts.

The maximum in and out position of the beam is sensed by a limit switch assembly consisting of aluminum tumblers which allow the roller drum to turn through 1,990° before actuating a double throw microswitch to control the motor.

The mount structure, which adapts the solar panel to the spacecraft is fabricated from 0.032 inch magnesium sheet. The part is designed as two closed intersecting torque boxes which serve to beam the load from adjacent panel assemblies to a cantilevered torque box which transfers the load to the spacecraft.

Figure 12 is a photograph of the solar cell panel fully deployed showing the rear surface of the substrate and the outboard end of the beam assembly.

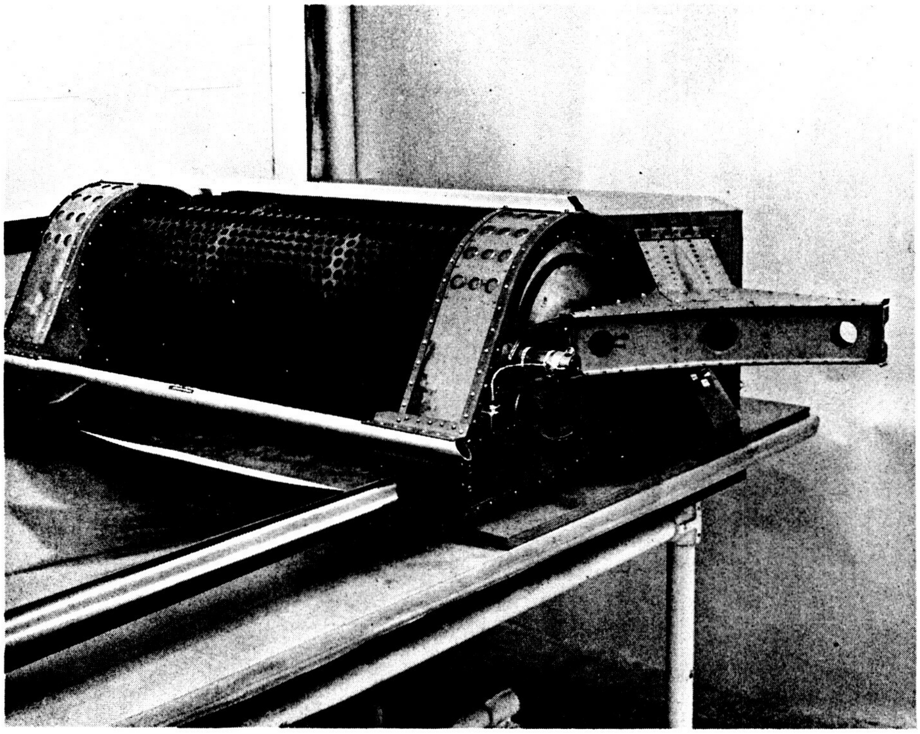


Figure 11. Solar Cell Array Structure

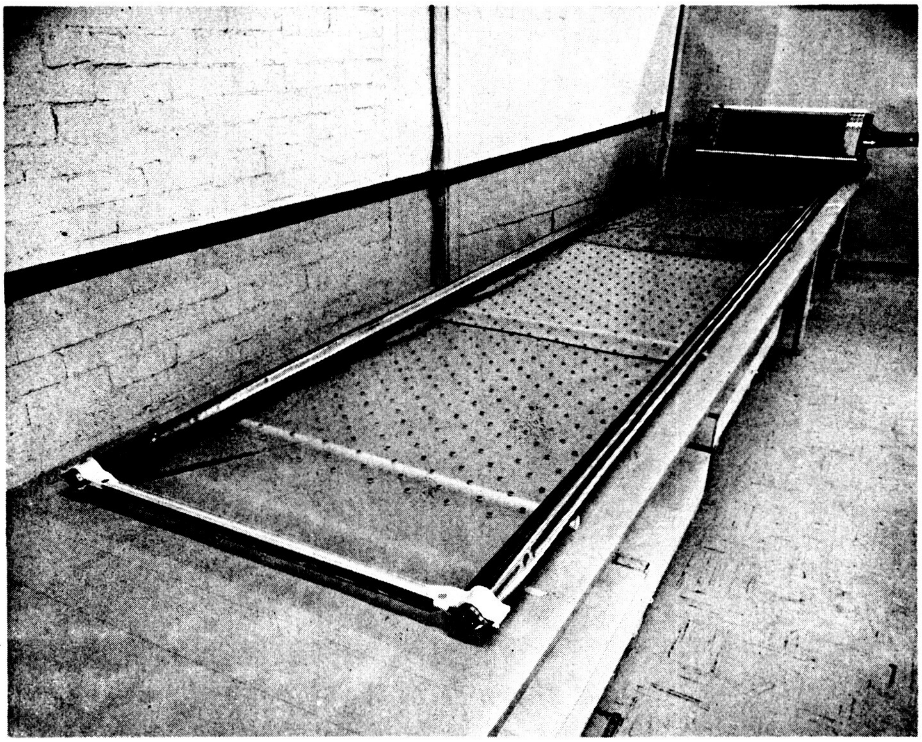


Figure 12. Solar Cell Panel - Deployed

The basic material of the substrate is epoxy resin impregnated fiberglass cloth. The substrate is 0.003 inch thick. The slotted edge is built up to 0.012 inch thickness. The ends of each substrate are provided with piano-type hinges which permit modules to be easily replaced.

The back surface of the substrate contains 1/2-inch diameter silicone foam rubber pads to cushion the solar cells. The pads are faced with 0.0002 inch thick teflon sheet. Four modules attached end to end provide 51 square feet of solar cells. Additional space is available for mounting cells by extending the substrate from the present end still further inboard to meet the drum. This could increase the area to 57 square feet.

The outside of the drum (Figure 13) is set by JPL specification to a minimum of one foot diameter. The drum consists of a cylindrical 0.025 sheet magnesium skin 43.1 inch long. Approximately 30% of the area of this skin is removed by lightening holes. An access door is provided for electrical harness access. At each end of the drum assembly is a honeycomb sandwich bulkhead. A magnesium hub is mounted at the center of each sandwich to support the static and dynamic loads of the drum assembly. These hubs act as an axle for rotation and transfer loads to the support structure. A magnesium spur gear is attached at the one end to provide the method of driving the drum. At the other end is a guide for the electrical harness.

Figure 14 shows the electrical harness is the deployed and retracted configuration.

The coiled harness permits rotation of the drum while routing the power from the panels through the structure. Eight #22 wires are attached to a fiberglass strip which serves as a spring carrier. This flat harness is spirally wrapped around a spool inside the drum assembly in a retainer, created by the structural bulkheads in the drum. One of the harnesses is terminated in an electrical connector secured to the drum. Access to this connector is obtained through the access door in the drum. It is here that the electrical feed from the cells is connected. The opposite end of the harness feeds through the spool and is secured to the panel assembly support structure. The spool is used to wind the spiralled harness to a compact coil for its stowed position during the boost phase. When the panel assembly is deployed, the rotating drum causes the tightly wound spiral to unwind gradually during the 5-1/2 turns of the drum. The harness always remains confined within the drum assembly and merely relaxes or tightens on the spool as the substrate deploys or retracts, this scheme precludes any need for a slip ring assembly.

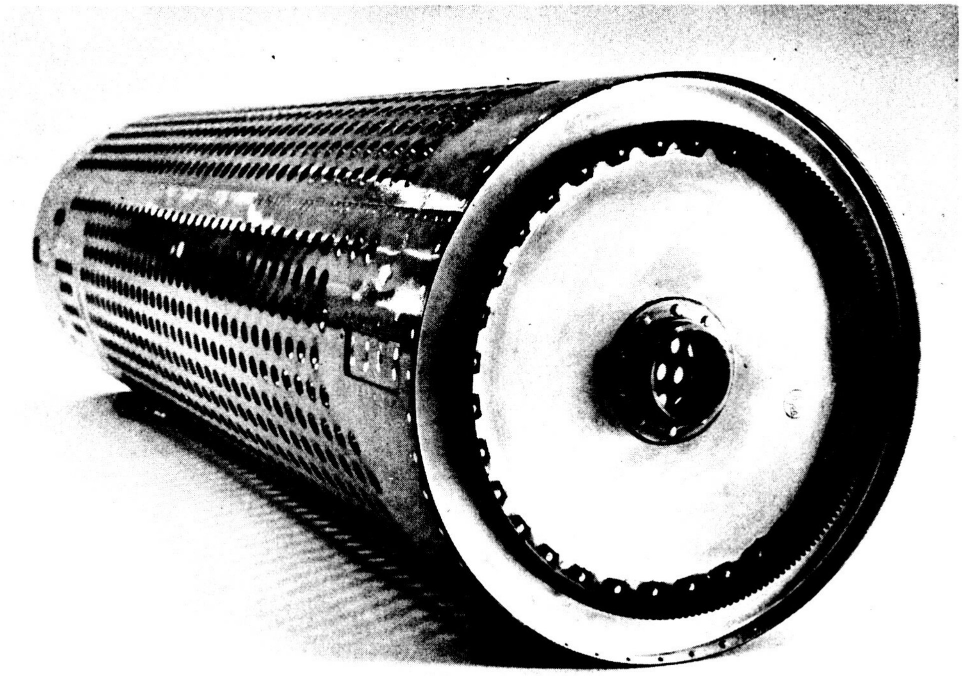


Figure 13. Storage Drum

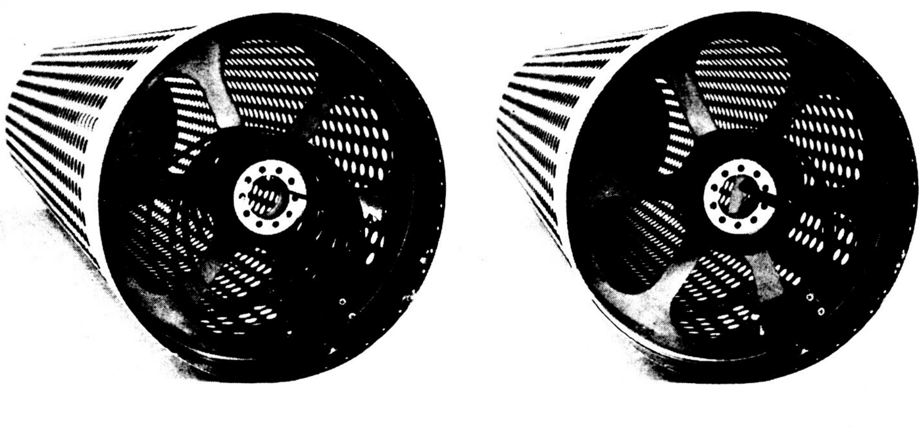


Figure 14. Electrical Harness - Deployed and Retracted

Figure 15 is a photograph of the support structure end assembly showing the internal construction.

The end assembly is a magnesium housing for the spring loaded rollers which restrain the beam. Mounting bosses are attached to the sides of the assembly for the installation of beam support guides. The structure provides for mounting the assembly on the spacecraft and for installation of the drive motor. This part of the assembly transfers loads, imposed by the extended beam, to two cross beams which tie the two assemblies together.

The limit switch tumbler assembly can be seen at the center of the bearing mount for the storage drum.

Figure 16 is a photograph of the beam guide sleeve.

The beam guides are shaped to the contour of the beam at the transisition area. These beam guides react loads imposed by the beam at full extension and also follow the changing position of the drum tangent point as successive layers are added or removed. The guide is pivoted at the outer end. A cross shaft which is rotated by the beam travel on a friction wheel causes rotation of a cam support and provides a means of reacting beam loads at any position along the total travel of the beams. The guide is fabricated from fiberglass reinforced epoxy resin and stiffened magnesium formers. The inner surface is lined with teflon.

4.4 DETAIL DESIGN

The design is shown in detail by Ryan drawings 208V001 through 208V012, and specifications 208S001 and 208S002. These drawings and specifications and additional detail information are described in the Final Design Report (Reference 2).

During manufacture of the prototype structure some minor changes were made to correct drawing errors or to facilitate manufacturing. Because of a limitation on availability of material, the inboard and outboard ends of the beam were made identical to permit an option of use as left or right hand beams. Aluminum rollers were substituted for magnesium in order to preclude purchase of special extrusion. Other minor variations were made in some of the assembly details to prevent interferences or to maintain design function. A list of drawings and revisions is presented in the following Table. The changes are discussed in Sections 5.0 and 7.0 of this report.

DRAWING LIST - JPL DEPLOYABLE SOLAR PANEL

Drawing No.	Revision	Title
208V001	ADI 343503	Deployable Panel Assembly
208V002		Solar Array Installation - Support Structure Deployable Solar Panel
208V003	ADI 343504	Drum Assembly - Deployable Solar Panel
208V004 Sht 1 & 2		Beam Guide Installation - Deployable Solar Panel
208V005		Substrate Assembly - Deployable Solar Panel
208V006	ADI 343502 ADI 343507 ADI 343508 ADI 343509	End Cap Assembly
208V007	A	Beam Installation - Deployable Solar Panel
208V008	ADI 343505	Harness Installation - Deployable Solar Panel
208V009		Gear-Drum Drive - Deployable Solar Panel
208V010	ADI 343506	Gear-Motor Drive - Deployable Solar Panel
208V011		Switch Installation - Deployable Solar Panel
208V012	A	Connector, Substrate - Deployable Solar Panel
208S001		Assembly and Adjustment Instructions - Deployable Solar Panel
2085002		Process Requirements for Deployable Solar Array - Deployable Solar Panel

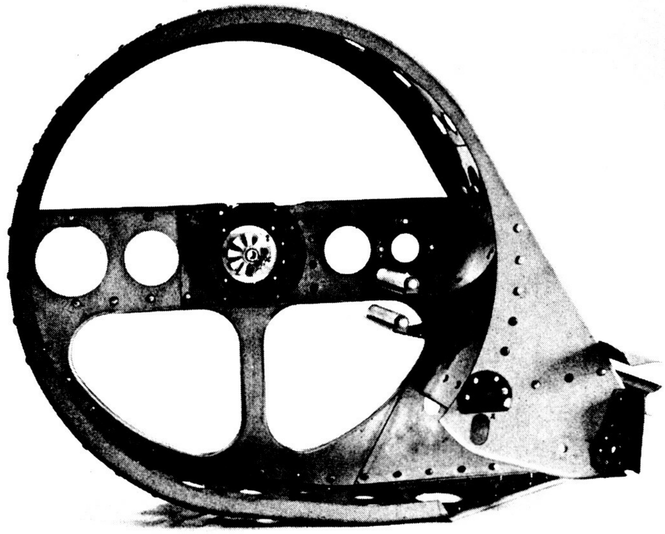


Figure 15. Support Structure End Assembly

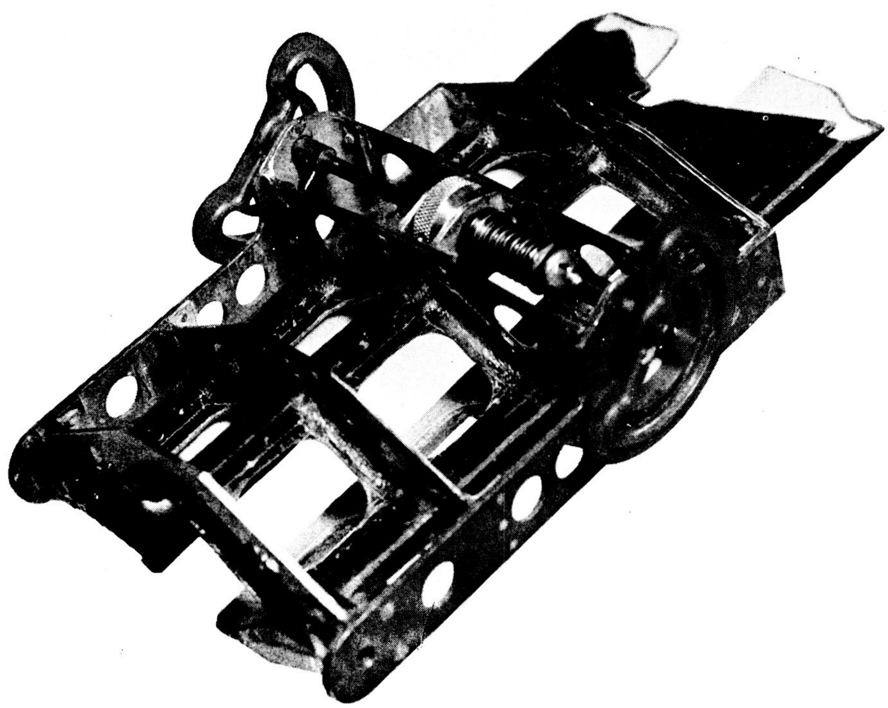


Figure 16. Guide Sleeve - Beam Support

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5.0 STRUCTURAL AND THERMAL ANALYSIS

A detailed analysis was performed to substantiate the design. Critical loading conditions for the design are the .2 g cruise maneuver load and the booster burnout dynamic load of 4.0 g rms at 20-200 cps.

The cruise maneuver load established the maximum bending moment on the deployed beam and controlled the beam strength and weight and consequently the end assembly spring rollers. When applied parallel to the spacecraft axis, the 0.2 g load causes the deployed substrate to deflect 17.0 inches at the tip. The booster burnout established the loads and strength requirements for the support structure.

The beams as designed will buckle on application of the retro-maneuver loads; therefore, the option of retracting the array during retro-maneuver was chosen to reduce weight.

Thermal gradients and steady state temperatures were determined for the deployed beams and substrate. Radiative surface properties of the beam and substrate materials were measured and a thermal mathematical model was created to determine representative temperatures for the array and beam.

The analysis is present in detail in the Final Design Report (Reference 2).

5.1 STRUCTURAL CONSIDERATIONS

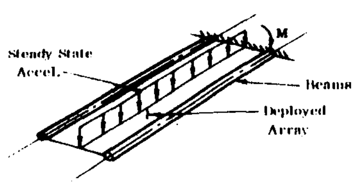
Consideration is given only to those primary structural areas which, for fabrication reasons or reasons of mill sheet thicknesses differing from that of drawing requirements, affect the structural characteristics of the design. These areas of concern presented are: (1) the deployable beams, (2) the substrate in stowed configuration, (3) the stowed beam radial restraint springs and (4) the structure weight; comparison is made with theoretical predictions presented in Final Design Report.

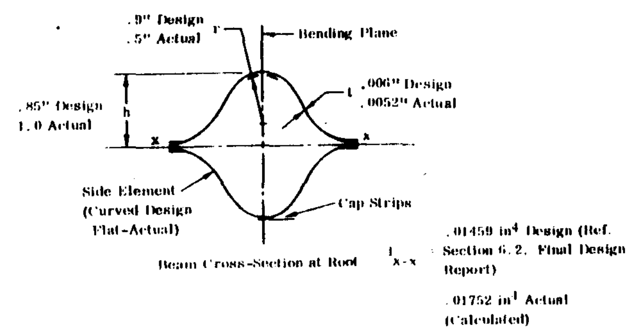
The adjusted theoretical beam allowables presented here will be used in the test phase of the program for correlation with test results.

5.1.1 Deployable Beams

Mill sheet thickness of the 6 AL-4V titanium was below the minimum drawing tolerance; actual sheet thickness was measured at 0.0052 inch compared to a required minimum of 0.006 inch. However, the actual beam depth was greater than design. The effects on the theoretical structure characteristics, given in the final design report, are presented as follows.

a. **Cantilevered Bending Capacity - The 0.2g steady state acceleration normal to the deployed array plane induces an equivalent room temperature bending moment of 422 in-lbs. ultimate per beam at the root section. (Reference Section 11.2, Final Design Report). The change in sheet thickness and cross-section geometry of the beam shown below affects the cantilevered normal-to-plane bending characteristic, M, as follows**





$$M_{ACTUAL} = M_{DESIGN} \cdot \left(\frac{tI}{rh}\right)_{ACTUAL} \cdot \left(\frac{rh}{tI}\right)_{DESIGN}$$

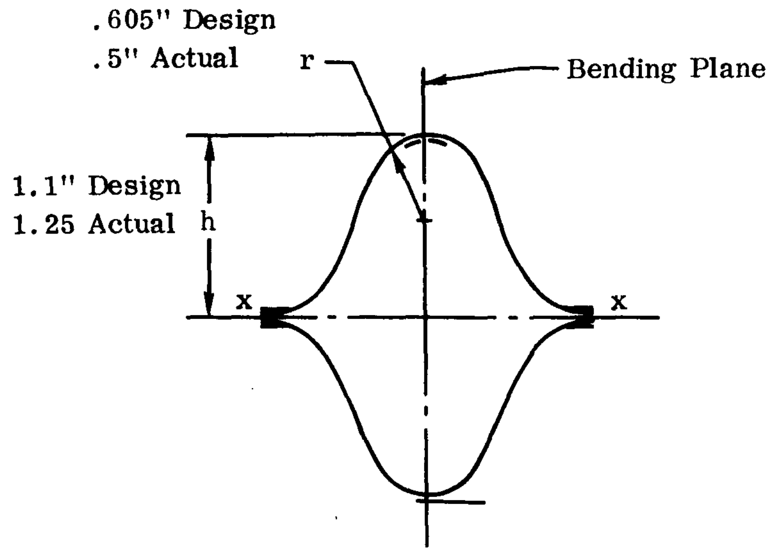
$$= 445 \times \left(\frac{.0052 \times .01752}{.5 \times 1}\right) \cdot \left(\frac{.9 \times .85}{.006 \times .01459}\right)$$

M_{ACTUAL} = 445 x 1.59 = 708 in-lbs. Ult., which corresponds to

a steady state acceleration of .2 x 1.59 = <u>.32 g.</u>

Elastic buckling of the flat sides will occur at approximately 61% (or 432 in. pounds) of the calculated ultimate, but since the loads are steady state and not dynamic, this does not constitute an ultimate failure.

b. Cantilevered Bending Frequency - The theoretical first cantilevered banding frequency of the deployed array structure is corrected for the actual beam cross-section geometry, taken at a full-open section shown below,



Beam Cross-Section Full-Open

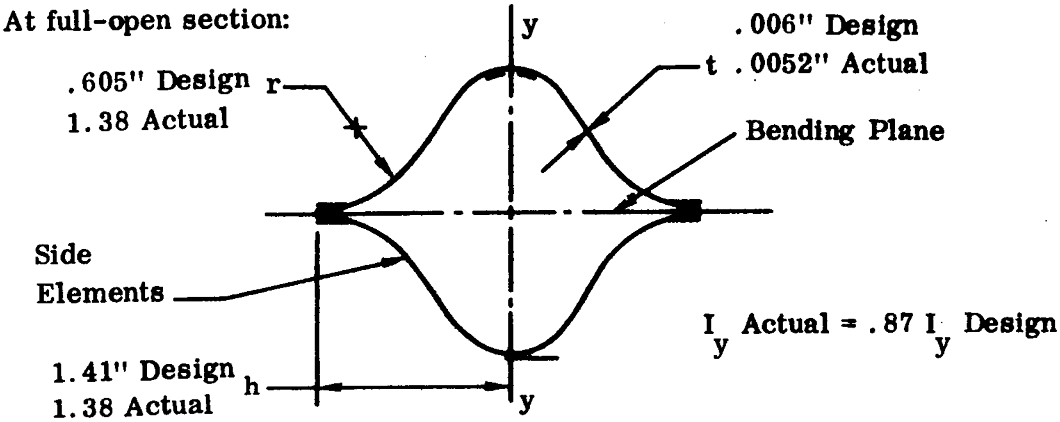
$$f_{n_{ACTUAL}} = f_{n_{DESIGN}} \cdot \left(\frac{I_{ACTUAL}}{I_{DESIGN}}\right)^{1/2}$$

$f_{n_{DESIGN}}$ = .55 cps at operating temperature,
.58 cps at room temperature
(Ref. Section 6.2, Final Design Report)

$$f_{n_{ACTUAL}} = .58 \cdot \left(\frac{.025}{.02}\right)^{1/2} = \underline{\underline{.65 \text{ cps}}}$$

c. Lateral Bending Capacity - The 0.2g normal-to-plane steady state acceleration on the deployed array induces an in-plane (lateral) bending moment in the beams as the substrate deflects normal to its plane. The beam lateral bending capacity at the full open cross-section and at the beam ends is corrected for the actual beam cross-section geometry and compared with that of 571 inch pounds ultimate required given in Section 5.11 of the Final Design Report.

At full-open section:



Beam Cross-Section
Full-Open

$$M_{ACTUAL} = M_{DESIGN} \cdot \left(\frac{tI}{rh}\right)_{ACTUAL} \cdot \left(\frac{rh}{tI}\right)_{DESIGN}$$

$$= 1132 \times \left(\frac{.0052 \times .87}{1.1 \times 1.38}\right) \cdot \left(\frac{.605 \times 1.41}{.006 \times 1}\right)$$

M_{ACTUAL} = 1132 x .420 = 475 in-lbs., which is less than required. However, this is an elastic buckling capacity of the side elements; the ultimate failure will occur at approximately 140% (665 in-lbs) of this calculated value.

$$M.S. = \frac{665}{571} - 1 = \underline{\underline{+.16}}$$

At beam ends:

$$M_{ACTUAL} \cong 1.4 \left[M_{DESIGN} \cdot \left(\frac{tI}{r}\right)_{ACTUAL} \cdot \left(\frac{r}{tI}\right)_{DESIGN}\right]$$

$$= 1.4 \left[1740 \times \left(\frac{.0052 \times .87}{1.3}\right) \cdot \left(\frac{.78}{.006 \times 1}\right)\right]$$

$$M_{ACTUAL} = 1.4 \left[787\right] = \underline{\underline{1102}} \text{ in-lbs}$$

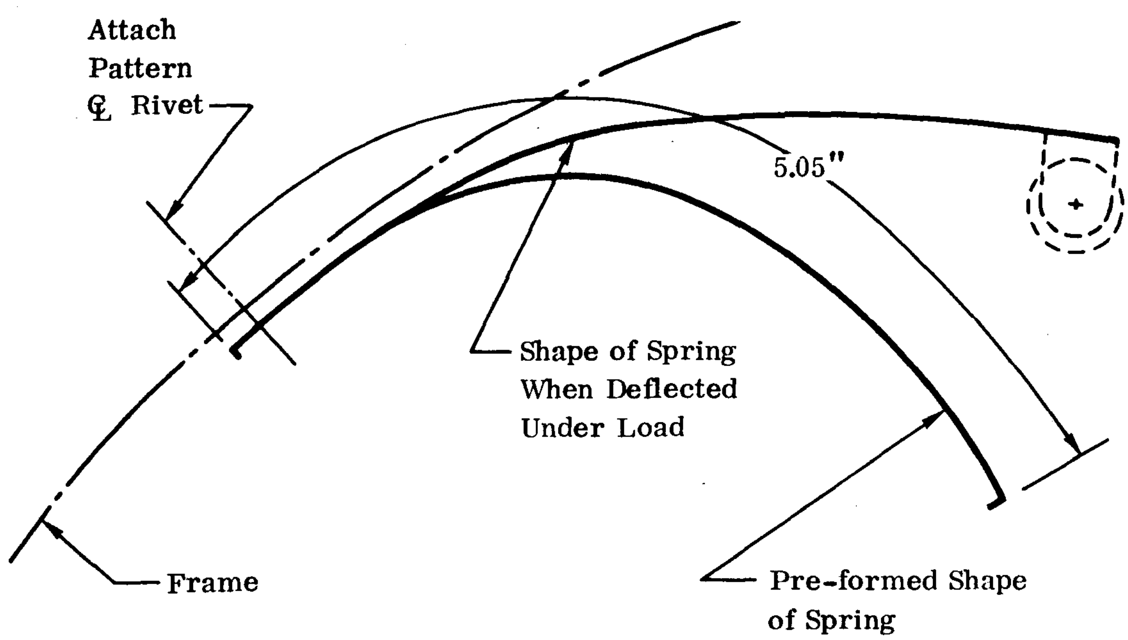
$$M.S. = \frac{1102}{571} - 1 = \underline{\underline{+.93}}$$

5.1.2 Stowed Substrate Dynamic Considerations

The outer layers do not wrap with such precision around the drum to justify a theoretical fundamental vibration frequency mode above the sinusidal excitation range (above 200 cps) as suggested in Section 5.8 of the Final Design Report; a radial breathing mode of vibration could occur in the outer layers. However, since it will be confined to the outer layers, dynamic radial pressure build-up is not sufficient to produce structural damage to the wrap drum or to the solar cells, by excessive deflection of the sponge pads allowing the solar cells to contact those on the inner wrapped layers. The effects on structural and electrical integrity of the solar cell installation due to vibration fatigue will be determined in the test phase of the program.

5.1.3 Stowed Beam Radial Restraint Spring Considerations

The radial restraint springs were designed to hold the flattened beams against the drum. A mockup was used early in the design phase to determine that a magnitude of approximately two pounds measured radially to the drum at each spring was required. Load-deflection verification tests conducted on a curved prototype spring per the design showed that the stiffeners were insufficient to satisfy the two pounds force requirement; the result was to increase the titanium spring sheet thickness and modify the design. A more efficient suggested redesign of the spring should involve shortening of its length to compensate for the increase in deflection due to its curvature.



5.1.4 Structure Weight

The calculated maximum and minimum weights (based on all sheet thicknesses maximum or minimum, respectively) are compared with actual item weights. The calculated weight summary shown is approximately 1.3 pounds greater than given in the final design report, which is accounted for by a few small items unintentionally omitted from that analysis. Mainly, the difference between the actual and calculated weights shown below is attributed to the increase in sheet thickness required for the stowed beam radial restraint springs (Part No. 208V006-43); resulting weight difference is approximately 1.6 pounds. The structure unit area weights given below are based on the actual available solar cell area of 51 ft.^{2}

WEIGHT SUMMARY/PANEL

Drawing Number	Assembly Name	Reqd.	Calculated		Actual
			Maximum	Minimum	
208V001	Solar Panel Assembly	1	2.9196	2.8274	2.6639
208V002	Support Structure *	1	2.0784	1.9650	2.0392
208V003	Drum Assembly	1	5.0267	4.6283	5.0308
208V004	Beam Guide Installation	1	1.5372	1.4082	1.5036
208V005	Substrate Assembly	1	2.3896	1.9441	2.3072
208V006	End Cap Assembly	1	4.2150	3.7060	6.5362
208V007	Beam Installation	1	5.7780	5.3120	4.8720
208V008	Harness Assembly *	1	0.3867	0.3669	0.3726
	TOTAL STRUCTURE		24.3312	22.1579	25.3255
	TOTAL LESS *		21.8661	19.8260	22.9137

Total Structure Weight/Unit Area of Solar Cells (lbs/ft^2)

Calculated		Actual
Maximum	Minimum	
0.4771	0.4345	0.4966

208V001 - SOLAR PANEL ASSEMBLY

Item	Dash No.	Reqd.	Calculated Maximum	Calculated Minimum	Actual
Motor Drive Assembly	-3	1	0.0362	0.0341	0.03196
Mount Plate	-5	1	0.0143		0.01433
Hat Section	-7	1	0.5450	0.5190	0.1850
Hinge Pin	-11	4	0.1040	0.0936	0.1190
Weld Assembly	-13	1	0.0103	0.0093	0.02094
Strut	-19 / -20	2	0.0088		0.0088
Cross Tie	-21	1	0.5677		0.5677
Drive Motor	—	1	0.7500		0.7562
Bearing	—	2	0.2600		0.2478
208V010 - Gear	—	1	0.0187		0.0187
208V006-41	—	1	0.0348	0.0314	0.04291
208V011-1 - Switch	—	1	0.1200		0.08350
208V012 - Connectors	—	124	0.2728	0.2232	0.123
Nuts, screws, rivets, adhesive and miscellaneous	—		0.1770		0.1441
	TOTAL (208V001)		2.9196	2.8274	2.6639

208V002 - SUPPORT STRUCTURE

Item	Dash No.	Reqd.	Calculated Maximum	Calculated Minimum	Actual
Web	-5	2	0.3400	0.3180	0.5048
Spar Channel	-7	2	0.4380	0.4120	0.4100
Closure Channel	-9	1	0.209	0.196	0.2061
Support Channel	-11	2	0.2520	0.2360	0.2513
Rib	-13	1	0.036	0.034	0.0573
Bracket	15	2	0.3440	0.3280	0.2183
Bracket	17	2	0.1620	0.1540	0.1014
Clip	-19	2	0.0140	0.0130	0.0198
Clip	-21	2	0.0194	0.0180	0.0132
Doubler	-23	2	0.1360	0.1280	0.1587
NAS 1003-4H Bolt Nutplates, Rivets, and Miscellaneous	—		0.1280		0.0983
	TOTAL (208V002)		2.0784	1.9650	2.0392

208V003 - DRUM ASSEMBLY

Item	Dash No.	Reqd.	Calculated Maximum	Calculated Minimum	Actual
Skin	-15	1	2.093	1.936	1.9885
Spacer	-17	2	0.0126	0.0114	0.0166
Doubler	-19	2	0.0252	0.0226	0.0220
208V009 - Gear	—	1	0.2741	0.2479	0.3329
Bulkhead	-23	1	0.0820	0.0759	0.0694
Door Skin	-25	1	0.0287	0.0265	0.0265
Spacer	-27	2	0.0062	0.0056	0.0034
Spacer	-29	2	0.0052	0.0048	0.0034
Spacer	-31	1	0.0154	0.0142	0.0165
Doubler	-33	1	0.0266	0.0246	0.0265
Bushing	-35	2	0.1960		0.1960
Bulkhead	-37	1	0.1374	0.1271	0.1279
Spool	-39	1	0.0557	0.0469	0.1014
Snap Ring	-41	1	0.0060	0.0054	0.0088
End Plate	-45	2	1.2286	1.1056	0.7892
Honeycomb	-47	2	0.0360	0.0342	0.0351
Skin	-49	2	0.7980	0.7436	0.6547
Miscellaneous					0.6120
TOTAL (208V003)			5.0267	4.6283	5.0308

208V004 - BEAM GUIDE INSTALLATION

Item	Dash No.	Reqd.	Calculated Maximum	Calculated Minimum	Actual
Guide Assembly	-3	1	0.7686	0.7041	0.7518
Guide Assembly	-4	1	0.7686	0.7041	0.7518
TOTAL (208V004)			1.5372	1.4082	1.5036

208V005 - SUBSTRATE ASSEMBLY

Item	Dash No.	Reqd.	Calculated Maximum	Calculated Minimum	Actual
Substrate	7	1	1.9588	1.6100	1.7901
Hinge	-9	7	0.0910	0.0714	0.1698
Hinge	-11	3	0.0390	0.0306	
Retainer	-13	1	0.0152	0.0130	0.02425
Stiffener	-15	1	0.0309	0.0253	0.02976
Damper Pad Assembly	-17	901	0.1747	0.1398	0.1982
Primer, Adhesive and Miscellaneous			0.0800	0.0540	0.0951
	TOTAL (208V005)		2.3896	1.9441	2.3072

208V006 - END CAP ASSEMBLIES

Item	Dash No.	Reqd.	Calculated Maximum	Calculated Minimum	Actual
End Plate	-5 / -6	2	0.1320	0.1200	0.2116
Angle	-7 / -8	2	0.1046	0.0974	0.0838
Angle	-9 / -10	2	0.1086	0.1014	0.0838
Former	-11	2	0.1920	0.1800	0.2050
Former	-13	2	0.1170	0.1096	0.1300
Doubler	-15	2	0.0936	0.0842	0.0889
Doubler	-17	2	0.0620	0.0580	0.0618
Angle	-19 / -20	2	0.3340	0.3160	0.3440
Former	-21 / -22	2	0.4220	0.4010	0.4144
Bracket	-23 / -24	2	0.0244	0.0230	0.0352
Bracket	-25 / -26	2	0.0110	0.0104	0.0286
Angle	-27 / -28	2	0.0300	0.0280	0.0440
Angle	-29	2	0.0360	0.0340	0.0540
Angle	-31	2	0.0194	0.0182	0.0110
Angle	-33 / -34	2	0.0150	0.0138	0.0110
Cover	-35 / -36	2	0.5680	0.5260	0.5732
Housing	-37 / -38	2	0.1832	0.1648	0.1786
Spool	-39	4	0.0296	0.0268	0.0441
Retainer	-41	1	0.0348	0.0314	0.04298
Spring	-43	44			2.6677
Bracket	-45	88			0.09699
Tube	-47	44			0.6301
Plug	-49	88	1.6368	1.2980	0.1936
Sleeve	-51	88			0.2908
Spacer	-53	2	0.0110		0.0110
Miscellaneous			0.0500		0.0500
	TOTAL (208V006)		4.2150	3.7060	6.5362

208V007 - BEAM INSTALLATION

Item	Dash No.	Reqd.	Calculated Maximum	Calculated Minimum	Actual
Beam Assembly	-5	1			1.95
Beam Assembly	-6	1			1.95
Cap	-7	4	4.0400	3.7360	—
Strip	-9	2	0.3340	0.3080	—
Plate	-11	2	0.0300	0.0280	0.0330
Plug	-15	2	0.0640	0.0600	0.0838
Channel	-17	2	0.2940	0.2640	0.300
Angle	-19	2	0.1040	0.0940	0.0926
Strip	-21	8	0.4080	0.3760	—
Strip	-23	8	0.0480	0.0400	—
Strip (Rubber)	-25	2	0.4260	0.3840	0.3510
Hinge	-27	1	0.016	0.014	0.02645
Adhesive, Screws, Rivets		1	0.010	0.008	0.0851
	TOTAL (208V007)		5.7780	5.3120	4.8720

208V008 - HARNESS ASSEMBLY

Item	Dash No.	Reqd.	Calculated Maximum	Calculated Minimum	Actual
Strip	-5	1	0.0562	0.0505	—
Cable	-7	1	0.0327	0.0294	—
Clamp	-9	1	0.0183	0.0146	0.0209
Lug	-11	2	0.0029	0.0027	0.0026
Tube	-13	1	0.0018	0.0017	—
Bracket	-15	1	0.0071	0.0066	0.0066
Molding	-17	1	0.0630	0.0600	0.0154
Molding	-19	1	0.0620	0.0590	0.0073
Strip	-21	2	0.0013	0.0010	—
Strip	-23	20	0.0014	0.0014	—
Connector - Electric	—	1	0.1400	0.1400	0.0121
Screws, Adhesive, etc.	—				
	TOTAL (208V008)		0.3867	0.3669	0.3726

5.2 THERMAL ANALYSIS

The thermal analysis conducted in Phase II is presented in detail in the Final Engineering Report (Reference 2). Thermal gradients and steady state temperatures were determined for the deployed beams and substrate. Radiative surface peoperties of the beams and substrate materials were measured and a thermal mathematical model was created to determine representative temperatures for the array and beam.

The design selected for the beam specified heat treat temperature of 1,300° F to achieve the desired thermal radiative properties. The rear surface of the beam was required to be polished for low emittance. Other surfaces were to have the natural 1,300° F oxide finish for high emittance.

Test results are tabulated below and compared to the previous data determined in Phase II

Surface

Titanium - 6 AL-4V	α_s	ϵ_n @ 80° F	ϵ_n @ 260° F
Polished	0.532	0.134	0.153
Blue Oxide (1,000° F)	0.776	0.159	0.185
* Brown Oxide (1,300° F)	0.769	0.302	0.355
Dust Blast Blue Oxide (1,000° F)	0.835	0.600	0.630
Dust Blast Brown Oxide (1,300° F)	0.891	0.613	0.651

* Selected in detail design

A comparison of emittance values indicates that the 1,000°F dust blast surface has significantly higher emittance than the smooth 1,300°F oxide. Also the rear surface of the beam can acquire the low emittance desired without polish when its original mill finish is oxided at 1,000°F. This eliminates the polishing operation required for a 1,300°F oxide.

Therefore, the beam design was changed to heat treat the beams at 1,000°F, with interior and front surfaces dust blasted. The test data are presented in Section 9.0

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6.0 MATERIALS

6.1 SELECTION OF MATERIALS

Materials used in the design have been chosen to meet the functional and environmental requirements. Although a large number of materials can meet these conditions, the special requirements of the deployable array concept have largely dictated specific materials choices.

One significant factor affecting material selection is the sterilization requirement of 108 hours at 145° C. During this procedure, the array structure will be in the stowed position in which many components will be strained from their natural shape. This precludes the use of many otherwise acceptable polymeric materials which would be expected to creep during sterilization. However, all materials which have been designated are heat resistant and capable of withstanding the sterilization procedure. It can also be noted that as a rule, the higher heat resistant polymers are more stable in the deep space environment than those of lower heat stability.

Nonmagnetic materials are used throughout the structure assembly except for the drive motor. Metals used in the structure are magnesium, aluminum, titanium, corrosion resistant steel, beryllium-copper and brass. The nonmetallic materials can be classified as epoxy, teflon, silicone, or glass.

Materials have been selected whose properties are well known and which can be processed within the state-of-the-art. Materials whose implementation would require research programs have been avoided.

Materials testing and process development have been conducted to establish properties or processes peculiar to the design requirements. This work is described in the Preliminary Development Report No. 20869-1 (Reference 2).

6.2 BEAM

Both metallic and nonmetallic beam materials were considered. Titanium was selected as the preferred beam material because of its superior wrapping properties and its stability while wrapped during the heat-sterilization process. Several glass fiber reinforced plastics were also found satisfactory, but are inferior to titanium.

Titanium (6AL-4V) was selected because of its high mechanical properties in the annealed state, weldability, and good fatigue properties. Forming of annealed titanium can be accomplished at up to 1,350° F without affecting mechanical properties. Beams were developed at 1,000°F forming temperature. However, in order to increase emittance, the surface was dust blasted to give thermal radiative properties similar to a 1,300°F oxide.

6.3 SUBSTRATE

The substrate material is a glass fiber reinforced epoxy resin system employing EPON 828 resin with DION RP-7A aromatic amine hardener. The glass reinforcement is type 113 woven fabric. This system has good stability in vacuum over a temperature range from -400° to 350° F. Substrate thickness is the factor controlling wrapping ability. The stiffness of the glass fiber laminate will increase only about 25% over a 250° F temperature drop.

6.4 CUSHIONING MATERIALS

In order to protect the solar cells during launch, a cushion arrangement is provided on the reverse side of the substrate. Because the cushion surface contacts the cells directly, teflon (TFE) film is selected to provide an inert, low friction bearing surface. The cushion is silicone rubber foam which has little change in properties from -100° F to 400° F.

The spacer stip along the deployable beam surface is AMS 3304 Silicone rubber. This stock is chosen for low compression set and negligible change in hardness in space-thermal environment.

6.5 ADHESIVES

Both epoxy and silicone adhesives are used in the array structure depending upon material types to be joined and design requirements. All flexible bonds are designed with Silastic 140 and A-4094 primer.

Rigid joints are designed with Shell Chemical Company EPON 934, EPON 956 or American Cyanamid-Bloomingdale FM-1000. Principle reason for selection of EPON 934 is its good mechanical properties at 300° F needed to resist sterilization envionment and to withstand deep space environment.

6.6 SURFACE FINISHES

Consideration of surface finishes has dealt primarily with thermal control coatings and corrosion protection. In order to provide minimum weight, the use of paints to control temperature has been avoided. The titanium beam, for example, is allowed to oxidize sufficiently during forming to produce acceptable thermal radiative properties.

The aluminum substrate edge connector is coated with Cat-A-Lac 463-1-8 epoxy. The α/ϵ ratio of this paint is similar to the fiberglass substrate and will maintain connector temperature within the same range.

Magnesium requires a protective coating to prevent corrosion in the pre-launch environment. Dow 17 anodize treatment is used throughout the design except in special cases where Dow 19 is employed.

The drive motor bearings and gear box are lubricated with Versilube G-300 silicone grease.

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7.0 FABRICATION OF PROTOTYPE

7.1 FABRICATION PROCEDURE

The fabrication of the prototype structure has closely followed the design as described. The principle manufacturing task was developing the tooling and methods for forming the 20 foot beams.

There was a weight increase of less than 0.50 pound due to manufacturing variations. The major weight difference between the design and actual structure was caused by the increase in gauge of the retaining springs.

The materials and methods used were within the present state-of-the-art. All process and assembly procedures were straightforward except the development of the beam forming method. The detail procedures are specified in Process Specification 208S002.

7.2 BEAM DEVELOPMENT

The beam process developed during the Phase I program consists of holding the beam to the desired contour by means of internal and external tools while the titanium is annealed at 1,000° F to 1,300° F in an air atmosphere furnance. (Reference 1). In addition to the difficulty of inserting and removing the internal tool, this method prevents the formation of the desired uniform oxide coating on the beam surfaces. Therefore, a modified forming process was developed using lateral compressive forces against the beam flange. The resulting cross section is very close to the designed shape.

A forming temperature of 1,000° F was selected to produce a natural rear surface emittance equivalent to the polished surface originally chosen. The internal surface emittance was achieved by dust blasting the surface prior to welding and forming.

The tooling arrangement is shown in Figure 17. A 19-9DL steel heat treat fixture is supported on a steel box beam. The welded beam assembly is cleaned, compressed to shape within the heat treat fixture, and heated to 1,000° F to anneal the titanium and set the shape. Some problems were encountered in welding and forming because of a longitudinal bow which existed in the flat foil stock received from the material supplier. Figure 18 shows the amount of curvature in the

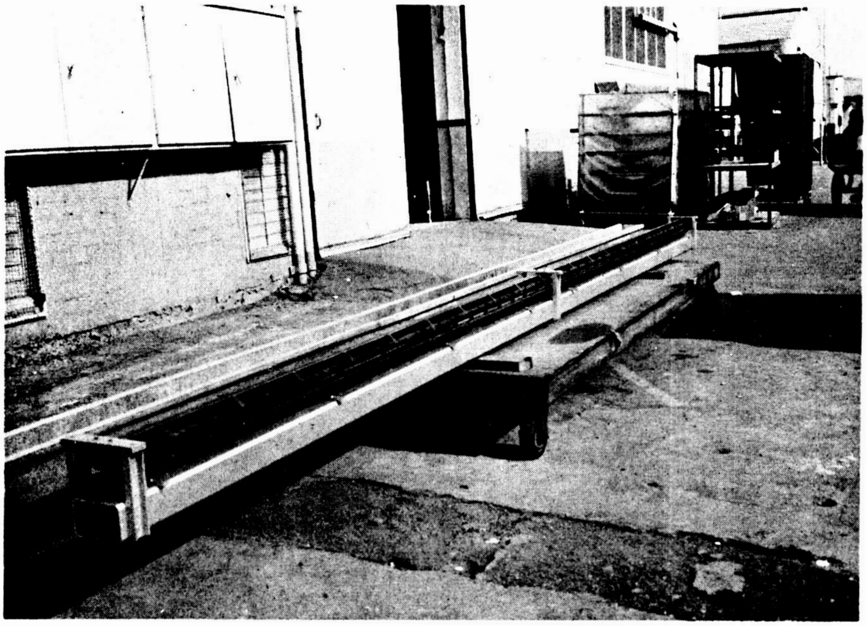

Figure 17. Beam Heat Treat Fixture

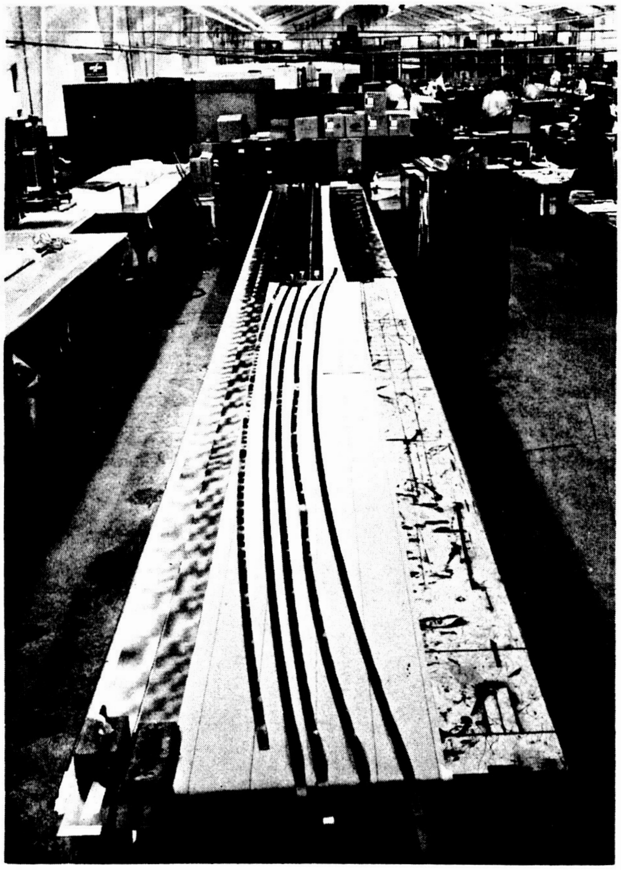

Figure 18. Raw Material for Beams

flattened foil. The attach flange material was severely curved. It was necessary to cut slots approximately every ten inches to provide a suitable edge for welding to the beam surface.

The two 20-foot beams are shown in Figure 19. The end of beam No. 2 was damaged in handling (Figure 20). The damaged area was trimmed away shortening the extended length of the array approximately two inches.

7.3 DEPLOYABLE ARRAY STRUCTURE ASSEMBLY

Assembly of the deployable array structure was generally straightforward with only minor assembly problems. The most severe problems encountered involved application of Dow 17 coating on the spot welded magnesium beam assembly. The drum assembly was corroded beyond an acceptable level at a vendor's facility because of entrapment of cleaner residues between the weld faying surfaces. Subsequently, a replacement drum assembly was manufactured and treated with Dow 19 and Cat-A-Lac 473-1-500 epoxy coating.

Figure 21 shows the prefit of the major components of the structure assembly. Some interferences were noted and rework was accomplished as requred to provide proper function of the assembly. During calibration of the beam assemblies it was necessary to match the longitudinal travel of each beam to prevent racking of the substrate during deployment and retraction. This was accomplished by reworking the height of the silicone rubber spacer strips and removing 40 inches from the inboard end of the strips.

One of the fiberglass substrate modules was reworked by cutting and splicing to provide a correct match of attach slots to the beam attach flange slots. Additional repairs were made to several substrate areas which were damaged during handling and in functional tests. A summary of minor variations during manufacturing is included in the Appendix, Section 9.0.

After assembly of the structure was completed, the substrate was covered with a 0.050-inch thick cardboard sheet scored in a one-inch grid to approximate the stiffness of solar cells during deployment and retraction cycles. The prototype structure has been successfully demonstrated in both vertical and horizontal deployment. A total of 34 deployment cycles have been completed in functional testing and demonstration. The power required for upward extension in a 1g field is 15 watts; the power required for horizontal deployment is 7.5 watts; for retraction, 10 watts.

The completed assembly is shown extended verically in Figure 22. The stowed configuration is shown in Figure 23.

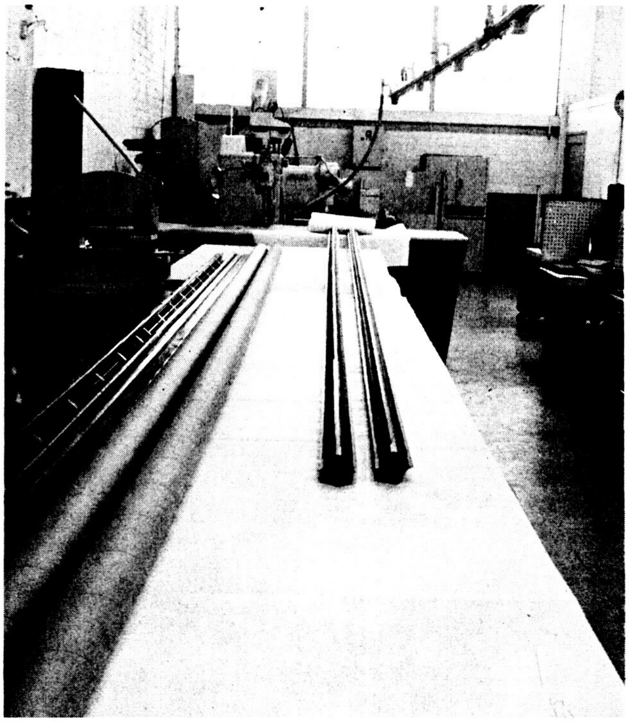

Figure 19.
20-Foot Beams
#1 and #2

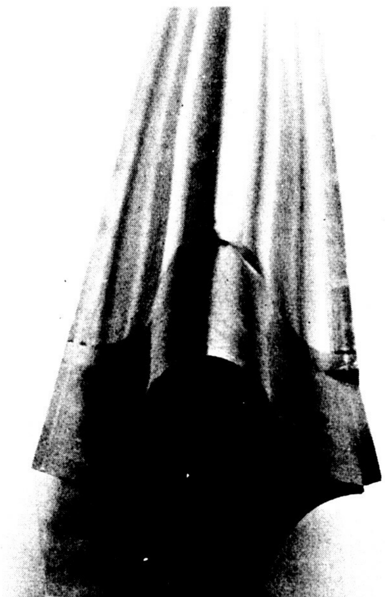

Figure 20.
Beam #2 - Damaged
End

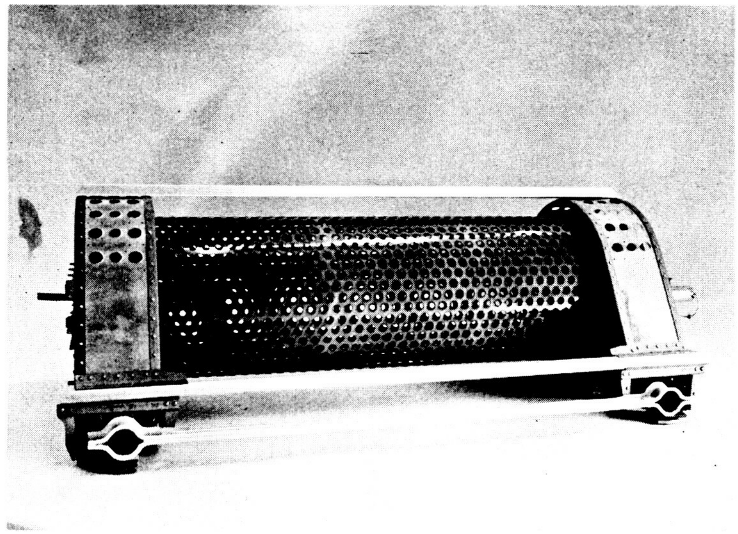

Figure 21.
Deployed Assembly
Prefit

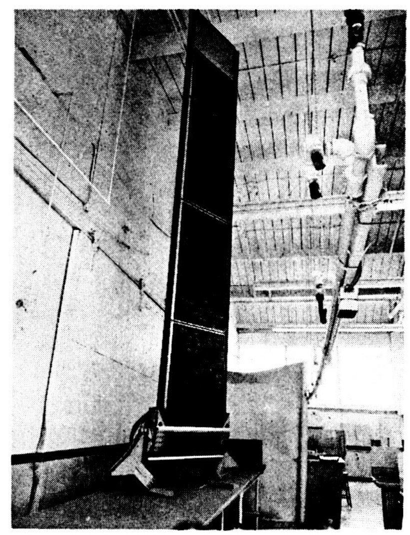

Figure 22.
Solar Array -
Vertically Extended

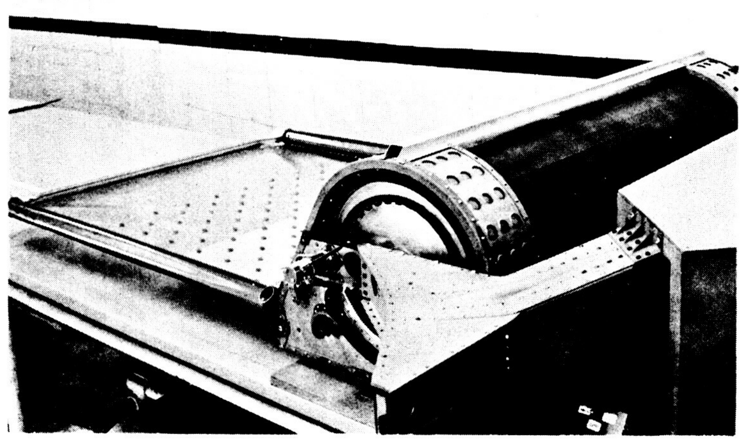

Figure 23.
Solar Array -
Stowed Configuration

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8.0 RECOMMENDATIONS

The manufacture of the prototype solar array structure was monitored to determine design and fabrication process recommendations which would improve function or facilitate production of the panel assemblies. The suggestions presented below are those which should be given consideration prior to production of flight panels.

8.1 DESIGN

The following design changes are recommended to improve function, reduce weight or facilitate production.

a. Redesign leaf springs in end assembly to reduce spring length and increase spring constant. This change will reduce the total assembly weight.

b. Modify limit switch to allow adjustment of travel in each direction. This will simplify calibration of beam travel after assembly.

c. Provide additional clearance between substrate edge connectors and substrate attach slots. This will permit more longitudinal self adjustment in the substrates to prevent substrate buckling due to variation in beam travel when wrapped.

d. Consider elimination of the guide cam mechanism to reduce weight. Operation of the prototype indicates that the movable guide may not be required.

e. Consider a sprocket drive system to provide positive control of beam travel during deployment and retraction.

f. Provide an access door in the drum assembly near the limit switch to allow adjustment. Also provide door in support structure for access to switch.

g. Redesign guide sleeves to allow removal or installation of beams without complete disassembly of the structure.

h. Extend the substrate inboard to fasten to the wrap drum. This will prevent damage when retracted during functional check. Also, this added area can be used for solar cells.

8.2 FABRICATION PROCESS

The fabrication processes developed for the prototype array are generally applicable to production fabrication. The following suggestions are made to simplify fabrication or reduce cost.

a. Provide assembly jigs to locate and control the beam installation to the wrap drum. Provide additional tooling to coordinate attachment slots in the beam and substrates.

b. Provide handling equipment and support fixture for the beam details and assemblies.

c. Consider a modification of the beam forming process to allow continuous forming rather than a batchwise process. This would require the development of new equipment but would provide a less cumbersome process at lower unit cost for a production quantity.

8.3 TEST PROGRAM

The prototype solar array structure has been designed and fabricated to meet the environmental and performance requirements of the JPL statement of work. In order to complete the evaluation of the design, it is recommended that a non-destructive test program be conducted to compare the prototype structure with design requirements. This program should evaluate the effects of sterilization, launch vibration, thermal loads, vacuum, and course correction maneuver on electrical and mechanical characteristics of the array.

9.0 APPENDIX

9.1 THERMAL RADIATIVE DATA

The thermal radiative data presented in this section were obtained using the procedures described in the Final Design Report, Paragraph 7.1.3 (Reference 2).

9.2 SUMMARY OF MANUFACTURING VARIATIONS

The following is a summary of variations from design which were required during manufacturing to complete fabrication of the prototype structure.

Description	Variation
(1) 208V007-5 Beam B/P 208V007 Zone 4C	Removed 40 inches of -25 strip from inboard end of -5 beam to calibrate rate of wind
(2) 208V005-65 Roller B/P 208V004 Zone 22D	Diameter of -65 was increased to 1.000. Because roller would not contact beam properly.
(3) 208V002-3 Support Structure	Four (4) 1-1/2 inch access holes added.
a. 208V002-11 Support B/P 208V002 Zone CID	a. One (1) 1-1/2 inch hole added 1-3/4 inches from outboard end and on centerline. 2 pieces required.
b. 208V002-9 Closure	b. Two (2) 1-1/2 inch holes added. 1-3/4 inches from each end and on center line.
(4) 208V007 Beam Installation -11 Plate should be installed on inboard end of -5 and -6 beams B/P 208V007 Zone 4C	-11 plate installation on outboard end of -5 and -6 beams. To correct the error, 0.600 material was removed from outboard end of -5 and -6 beams and the -11 plate was installed on the inboard end of -5 and -6 beams.

SUMMARY OF MANUFACTURING VARIATIONS (Continued)

Description	Variation
	-6 Beam has 3 cracks on upper surface of inboard end. 1 crack 1/4 inch in from edge, 0.100 long. 1 crack 1/2 inch in from edge 0.200 long 1 crack 3/4 inch in from edge, 0.090 long -11 plate was installed over the cracked area.
(5) 208V011 Switch Installation B/P 208V011 Zone 2B NAS 1351C06-4 Screw 4 required MS 35337-98 Washer 4 required	NAS 1190C6P6 Screw - 4 pieces
(6) MS 45525-4262 Snap Ring B/P 208V001 Zone 8B	Too wide to fit groove in 208V006-37 hub. Replaced by Ryan Manufacture Snap Ring 7075 T6 Material
(7) 208V001-3 Switch Assembly B/P 208V011	JX-40 actuator not a purchased part. Was manufactured in Ryan shop. 0.010 Berryllium copper. H.T. to 160,000-200,000 psi Rockwell C38
(8) 208V006-41 Retainer B/P 208V006 Zone B4	Right flange cracked from edge of hole to outside of material. Due to excessive clamp pressure Dow 17 processing.
(9) 208V003-3 Spotweld Assembly B/P 208V003 Zone A7 208V003-15 Skin See Flag Note 9	Process changed to Dow 19

SUMMARY OF MANUFACTURING VARIATIONS (Continued)

	Description	Variation
(11)	208V008-900 Harness Assembly B/P 208V008 Zone 6D 208V008-5 strip should be one continuous unbroken piece.	208V008-5 stip is cut and notched into approximately 20 separate strips to permit coiling.
(12)	208V011-3 Switch Assembly B/P 208V011 Zone 4A 208V011-7 Bracket 1 required	Two - 7 brackets used 160° apart. Engineering error.
(14)	208V006-3 Spring Assembly B/P 208V006 Zone 10A Five holes for 1/16 rivets should be dimpled far side	Five holes not dimpled. Changed rivets from MS 20426B-2 to MS 20470B-2.
(15)	MS 16624-4206 Snap Ring B/P 208V001-1 7one 8B	Due to inaccessable area, forbidding the installation of MS 16624-4206 Snap Ring, four 4-40 set screws were used to hold B545A151440-C bearing on 208V003-35 bushing in place of MS 16624-4206 Snap Ring
(17)	208V003-7 End Plate Assembly B/P 208V003 Zone 12C MS 20426B4 Rivet 10 places	Used MS20470A4-16 Rivet 10 places
(19)	208V005-15 Stiffener B/P 208V005 Zone 3A	208V005-15 Stiffener was removed from 208V005-5 Substrate and replaced with a fiberglass doubler bonded in place.
(20)	208V005-5 Substrate B/P 208V001 Zone C1	208V005-5 damaged on both corners of the inboard end approximately two inches long. Was repaired with EPON 956 and glass cloth.

REFLECTIVITY DATA RADIANT ENERGY TRANSFER GROUP SPACE SCIENCE LABORATORIES GD-ASTRO

RADIANT ENERGY TRANSFER GROUP SPACE SCIENCE LABORATORY GD-ASTRO

SAMPLE IDENT 6AL-4VII 1300 F DUST BLASTED DATE 7-29-66 REQUESTOR RYAN

Wave Length	Reflectance	Wave Length	Reflectance	Wave Length	Reflectance	Wave Length	Reflectance
0.300E 00	0.650E-01	0.320E 00	0.650E-01	0.330E 00	0.640E-01	0.335E 00	0.650E-01
0.350E 00	0.660E-01	0.360E 00	0.670E-01	0.370E 00	0.680E-01	0.375E 00	0.690E-01
0.390E 00	0.700E-01	0.400E 00	0.710E-01	0.410E 00	0.720E-01	0.420E 00	0.730E-01
0.430E 00	0.750E-01	0.440E 00	0.780E-01	0.445E 00	0.800E-01	0.450E 00	0.890E-01
0.465E 00	0.830E-01	0.475E 00	0.840E-01	0.490E 00	0.880E-01	0.495E 00	0.890E-01
0.500E 00	0.890E-01	0.510E 00	0.900E-01	0.520E 00	0.920E-01	0.540E 00	0.960E-01
0.570E 00	0.105E 00	0.700E 00	0.115E 00	0.800E 00	0.122E 00	0.900E 00	0.122E 00
0.100E 01	0.121E 00	0.110E 01	0.121E 00	0.120E 01	0.125E 00	0.130E 01	0.129E 00
0.140E 01	0.126E 00	0.150E 01	0.132E 00	0.160E 01	0.138E 00	0.170E 01	0.138E 00
0.180E 01	0.130E 00	0.200E 01	0.130E 00	0.250E 01	0.120E 00	0.300E 01	0.115E 00
0.350E 01	0.130E 00	0.400E 01	0.160E 00	0.450E 01	0.170E 00	0.500E 01	0.200E 00
0.550E 01	0.225E 00	0.600E 01	0.245E 00	0.650E 01	0.275E 00	0.700E 01	0.290E 00
0.750E 01	0.320E 00	0.800E 01	0.310E 00	0.850E 01	0.300E 00	0.900E 01	0.285E 00
0.950E 01	0.300E 00	0.100E 02	0.330E 00	0.105E 02	0.305E 00	0.110E 02	0.335E 00
0.115E 02	0.320E 00	0.120E 02	0.320E 00	0.125E 02	0.325E 00	0.130E 02	0.335E 00
0.140E 02	0.370E 00	0.150E 02	0.375E 00	0.160E 02	0.440E 00	0.170E 02	0.440E 00
0.180E 02	0.430E 00	0.190E 02	0.450E 00	0.200E 02	0.490E 00	0.210E 02	0.505E 00
0.220E 02	0.510E 00	0.230E 02	0.460E 00	0.240E 02	0.470E 00	0.250E 02	0.500E 00
0.260E 02	0.525E 00	0.270E 02	0.520E 00	0.280E 02	0.560E 00	0.290E 02	0.520E 00
0.300E 02	0.525E 00	0.310E 02	0.535E 00	0.320E 02	0.530E 00	0 .	0

EMISSIVITY REQUIRED 100 x 300 x 500 x SOLAR ABSORPTIVITY x OTHER
200 x 400 x CARBON ARC ABSORPTIVITY

EMISSIVITY (100 K) = 0.479693E 00 SUMMATION RATIO = 0.164646E 00

EMISSIVITY (300 K) = 0.613528E 00 SUMMATION RATIO = 0.567142E 00

EMISSIVITY (500 K) = 0.693362E 00 SUMMATION RATIO = 0.670396E 00

SOLAR ABSORPTIVITY = 0.891321E 00 SUMMATION RATIO = 0.891321E 00

EMISSIVITY (200 K) = 0.546129E 00 SUMMATION RATIO = 0.425079E 00

EMISSIVITY (400 K) = 0.651131E 00 SUMMATION RATIO = 0.612341E 00

REFLECTIVITY DATA RADIANT ENERGY TRANSFER GROUP SPACE SCIENCE LABORATORIES GD-ASTRO

RADIANT ENERGY TRANSFER GROUP SPACE SCIENCE LABORATORY GD-ASTRO

SAMPLE IDENT 6AL-4VII 1000 DUST BLASTED DATE 7-29-65 REQUESTOR RYAN

Wave Length	Reflectance	Wave Length	Reflectance	Wave Length	Reflectance	Wave Length	Reflectance
0.300E 00	0.125E 00	0.320E 00	0.130E 00	0.330E 00	0.134E 00	0.335E 00	0.137E 00
0.350E 00	0.141E 00	0.360E 00	0.146E 00	0.370E 00	0.148E 00	0.375E 00	0.150E 00
0.390E 00	0.152E 00	0.400E 00	0.154E 00	0.410E 00	0.155E 00	0.420E 00	0.156E 00
0.430E 00	0.157E 00	0.440E 00	0.158E 00	0.455E 00	0.158E 00	0.450E 00	0.158E 00
0.465E 00	0.158E 00	0.475E 00	0.158E 00	0.490E 00	0.158E 00	0.495E 00	0.158E 00
0.500E 00	0.158E 00	0.510E 00	0.158E 00	0.520E 00	0.158E 00	0.540E 00	0.158E 00
0.570E 00	0.159E 00	0.700E 00	0.160E 00	0.800E 00	0.161E 00	0.900E 00	0.160E 00
0.100E 01	0.161E 00	0.110E 01	0.166E 00	0.120E 01	0.172E 00	0.130E 01	0.175E 00
0.140E 01	0.175E 00	0.150E 01	0.175E 00	0.160E 01	0.180E 00	0.170E 01	0.186E 00
0.180E 01	0.185E 00	0.200E 01	0.200E 00	0.250E 01	0.215E 00	0.300E 01	0.220E 00
0.350E 01	0.235E 00	0.400E 01	0.260E 00	0.450E 01	0.260E 00	0.500E 01	0.275E 00
0.550E 02	0.285E 00	0.600E 01	0.295E 00	0.650E 01	0.310E 00	0.700E 01	0.315E 00
0.750E 01	0.325E 00	0.800E 01	0.330E 00	0.850E 01	0.335E 00	0.900E 01	0.330E 00
0.950E 01	0.330E 00	0.100E 02	0.355E 00	0.105E 02	0.330E 00	0.110E 02	0.350E 00
0.115E 02	0.330E 00	0.120E 02	0.335E 00	0.125E 02	0.335E 00	0.130E 02	0.345E 00
0.140E 02	0.365E 00	0.150E 02	0.395E 00	0.160E 02	0.410E 00	0.170E 02	0.450E 00
0.180E 02	0.425E 00	0.190E 02	0.450E 00	0.200E 02	0.500E 00	0.210E 02	0.525E 00
0.220E 02	0.515E 00	0.230E 02	0.475E 00	0.240E 02	0.470E 00	0.250E 02	0.500E 00
0.260E 02	0.530E 00	0.270E 02	0.515E 00	0.280E 02	0.545E 00	0.290E 02	0.525E 00
0.300E 02	0.520E 00	0.310E 02	0.535E 00	0.320E 02	0.540E 00	0. 02	0.

EMMISSIVITY REQUIRED 100 x 300 x 500 x SOLAR ABSORPTIVITY x OTHER
200 x 400 x CARBON ARC ABSORPTIVITY

EMISSIVITY (100 K) = 0.472652E 00 SUMMATION RATIO = 0.164308E 00

EMISSIVITY (300 K) = 0.599626E 00 SUMMATION RATIO = 0.554227E 00

EMISSIVITY (500 K) = 0.656213E 00 SUMMATION RATIO = 0.633736E 00

SOLAR ABSORPTIVITY = 0.834979E 00 SUMMATION RATIO = 0.834979E 00

EMISSIVITY (200 K) = 0.538440E 00 SUMMATION RATIO = 0.419966E 00

EMISSIVITY (400 K) = 0.629633E 00 SUMMATION RATIO = 0.591669E 00

10.0 LIST OF REFERENCES

1. Preliminary Development Report - Deployable Large Area Solar Array Structure. Report No. 20869-1, Ryan Aeronautical Company, Lindbergh Field, San Diego California 92112, 30 July 1965.

2. Final Design Report for Deployable Large Area Solar Array Structure. Report No. 20869-2, Ryan Aeronautical Company, Lindbergh Field, San Diego, California 92112, 5 May 1966